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Rotorcraft Noise

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Industry Workshop on Aerodynamic Noise
Prediction/Noise Reduction held at
Langley Research Center
Hampton, Virginia
March 29-31, 1983

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PREFACE

The papers included herein were presented at the NASA/U.S. Helicopter Industry Workshop on Aerodynamic Noise Prediction/Noise Reduction held at Langley Research Center, March 29-31, 1982. The workshop included prepared papers on the subjects of noise reduction techniques, scaling laws, empirical noise prediction, psychoacoustics, and methods of developing and validating noise prediction methods. Working sessions allowed the participants to enumerate their findings, conclusions, and recommendations with regard to the future direction of rotorcraft noise research.

The objective of the workshop was to establish realistic plans for NASA and the U.S. helicopter industry to develop a design-for-noise methodology, including plans for the identification and development of promising noise reduction technology. As such, the findings, conclusions, and recommendations of the workshop are [illegible] to provide guidance and justification for noise and aerodynamics research within NASA and the industry for the foreseeable future.

This workshop was organized under the direction of the cochairmen, Homer G. Morgan, Chief, Acoustics and Noise Reduction Division, Langley Research Center, and Richard L. Long, Director, Army Structures Laboratory (AVRADCOM), Langley Research Center. Special recognition is due Dr. J. P. Raney of the Noise Prediction Branch, Acoustics and Noise Reduction Division, Langley Research Center, and Danny R. Hoad of the Army Structures Laboratory for their preparation in the planning and execution of the workshop.

Use of trade names or names of manufacturers in this report does not constitute an official endorsement of such products or manufacturers, either expressed or implied, by the National Aeronautics and Space Administration.

Robert J. Huston
Workshop Technical Secretary

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INTRODUCTION AND OBJECTIVES

Robert J. Huston
NASA Langley Research Center

A major objective of the NASA/U.S. Helicopter Industry Workshop on Aerodynamic Noise Prediction/Noise Reduction was the establishment of realistic plans for NASA and the U.S. helicopter industry to develop a design-for-noise methodology, including plans for the identification and development of promising noise reduction technology.

External noise prediction and noise reduction are not suprising goals for either NASA or the U.S. helicopter industry. What may be suprising is the joint manner in which this task is being approached. The first steps should include the establishment of a consensus between the technical experts of the industry and the government research laboratories on the state of the art of rotorcraft noise technology, and the identification of specific technology advances required.

Each of the three NASA centers has an ongoing research program applicable to rotorcraft noise reduction. The recommendations made at this workshop can be expected to have influence on the direction taken by these programs. A cooperative NASA-industry research program has been proposed by the American Helicopter Society, and it can be expected that, as respective government and industry tasks are defined within this program, the recommendations and priorities of this workshop will be considered as primary guidance. In recognition of recent budget limitations, a proposed rotorcraft noise program should give specific recognition to military requirements as well as to potential noise regulations and predicted economic impact.

The Rotorcraft Subcommittee of the NASA Aeronautics Advisory Committee has recommended that rotorcraft noise be NASA's first priority for enhancement because of the degree to which noise technology lags regulatory and operations needs. This committee also made recommendations regarding acoustic treatment of wind tunnels at Langley and Ames Research Centers, possible use of European wind tunnels for acoustic tests, and joint NASA-industry research programs (such as the one on noise). A new advisory group, the Government Working Group on Rotorcraft Noise Research, has been formed to make specific recommendations on the content of the noise program. This group has representatives from each of the NASA aeronautics centers, NASA Headquarters, AVRADCOM, the FAA, and the four major helicopter companies.

The participants of the workshop included key individuals from each of the NASA aeronautics research centers and the major U.S. rotary-wing manufacturers, as well as representatives of AVRADCOM and the Army Research Office. The skills and disciplines of these individuals are varied, but all have as a common interest the economic or cost-effective reduction of noise from rotorcraft. A number of these participants are in positions of authority and can influence or redirect the use of resources at their disposal to effect a concerted attack on noise. Some are responsible for the design process, and are sensitive to the real limitations on the designer when faced with conflicting requirements. Others are experts in the areas of acoustics, rotorcraft aerodynamics, psychoacoustics, and propulsion.

This workshop should establish the need for and the direction and details of future rotorcraft external noise research within the U.S., and should make recommendations on the part that NASA and industry should play in that research. The findings and conclusions of the workshop must be considered carefully because they will most assuredly be used as guidance and justification for noise and aerodynamic research within NASA and the industry.

SESSION I

NOISE REDUCTION TECHNIQUES

Chairman: H. K. Edenborough
NASA Ames Research Center

Vice-chairman: J. M. Drees
Bell Helicopter Textron

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ACOUSTIC DESIGN CRITERIA AND VALIDATION

Peter Arcidiacono
Sikorsky Aircraft Division of United Technologies Corp.
Stratford, Conn. 06601

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QUESTIONS ADDRESSED BY THE SIKORSKY PRESENTATIONS

There are three Sikorsky presentations at this Workshop. The questions being addressed by the three presentations are listed below. The first presentation emphasizes Questions 1-4. The second presentation by R. Schlegel speaks largely to Questions 3-6 with more detail where overlap exists. The third presentation by D. Jenney is concerned largely with Questions 6-8.

1. **WHAT IS THE HELICOPTER ACOUSTIC DESIGN PROBLEM?**
2. **WHAT ARE THE OPTIONS FOR ADDRESSING THE PROBLEM?**
3. **WHICH DESIGN APPROACH IS IN USE TODAY?**
4. **WHAT IS THE IMPACT OF THIS APPROACH?**
5. **HOW WELL IS THIS APPROACH DOING?**
6. **WHAT IS THE DESIRED DESIGN CAPABILITY?**
7. **WHAT IS THE STRATEGY FOR OBTAINING THIS CAPABILITY?**
8. **WHAT IS AN OBJECTIVE METHOD FOR MEASURING PROGRESS?**

Figure 1

OBJECTIVES OF THIS PRESENTATION

The specific objectives of this presentation are listed below. It is important to understand the basics of the general design process and the likely role that acoustic requirements will play. The options available to the acoustic designer in meeting these requirements are reviewed as is the particular approach being followed at Sikorsky at the present time. Finally, the impact of the current approach is presented.

- **GENERAL DESIGN PROCESS & CRITERIA**
- **ACOUSTIC DESIGN REQUIREMENTS**
- **OPTIONS AVAILABLE FOR ACOUSTIC DESIGN**
- **CURRENT METHOD AND IMPACT**

Figure 2

LIST OF SYMBOLS

GW	Aircraft gross weight, lb
h	Distance between aircraft and microphone, ft.
$M_{\Omega R}$	Mach number based on rotational speed, ΩR
$M_{1,90}$	Mach number based on advancing blade speed, $\Omega R + V$
R	Rotor radius, ft
V	Aircraft forward speed, ft/sec
ΩR	Rotor rotational tip speed, ft/sec

BASIC DESIGN CRITERIA

The basic criteria followed in design is simply stated. The customer requirements must be met. The design meeting the requirement must be competitive. There must not be an undue risk in the design so that technical commitments can be met within estimated budget and schedule. The acoustic design criteria obviously must meet all of these system requirements. Noise control concepts are selected that meet the requirement, are competitive and are of reasonable risk.

MEET ALL CUSTOMER REQUIREMENTS WITH A DESIGN THAT IS COMPETITIVE AND OF REASONABLE RISK

Figure 3

BASIC DESIGN PROCESS

This figure shows the principal elements involved in the design and development of a new helicopter. At any point in time a technology base exists. Contributing to this technology base are (a) experimental data at a component and system level (model and full scale), (b) analytic procedures (first principle, semi-empirical and empirical) and (c) personnel experience. Since design requirements inevitably tend to exceed the scope of technology base available to any one company, a key part of the process is the identification of risk and tests to reduce those risks to an acceptable level. Confirmation testing at the component and system level together with the solving of any residual development and service problems complete the cycle.

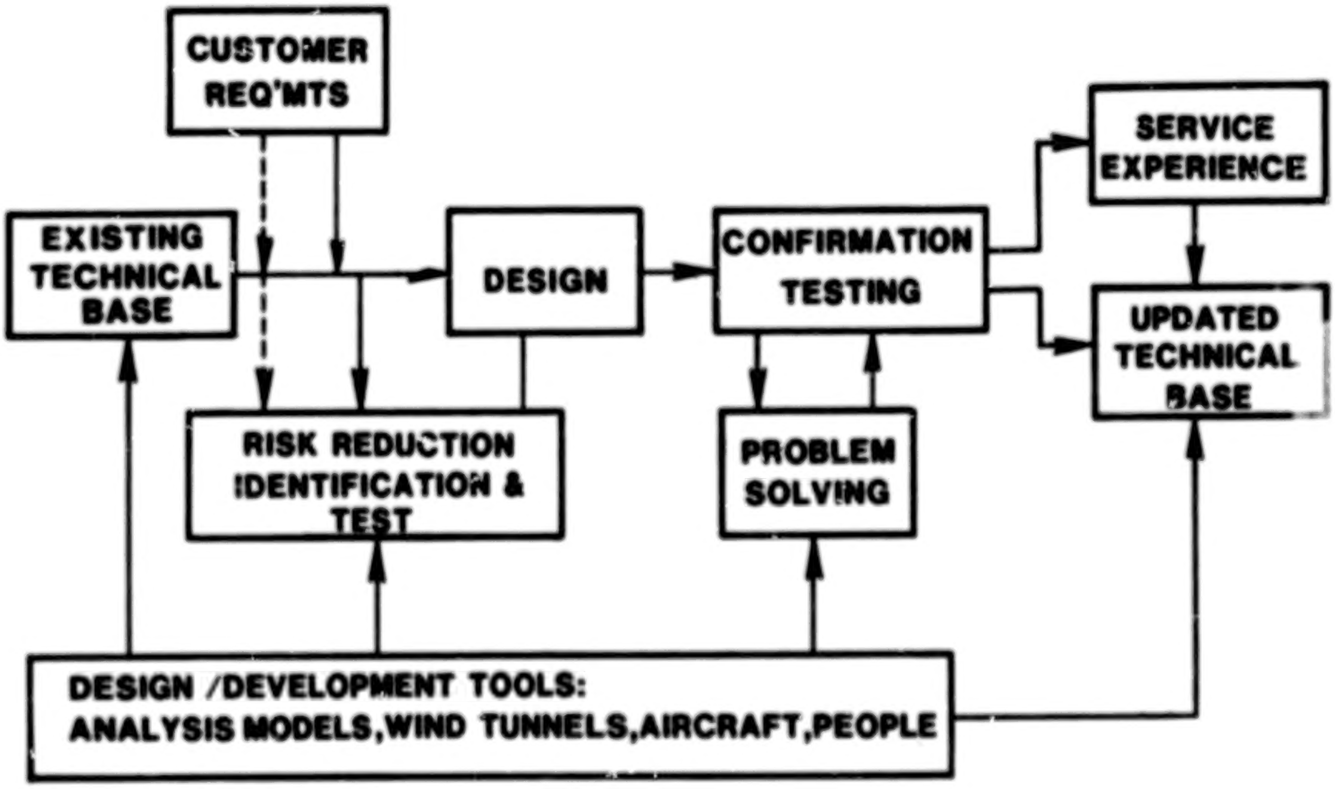
CUSTOMER REQ'MTS
EXISTING TECHNICAL BASE
DESIGN
CONFIRMATION TESTING
SERVICE EXPERIENCE
UPDATED TECHNICAL BASE
RISK REDUCTION IDENTIFICATION & TEST
PROBLEM SOLVING
DESIGN /DEVELOPMENT TOOLS:
ANALYSIS MODELS,WIND TUNNELS,AIRCRAFT,PEOPLE

Figure 4

SUCCESS REQUIRES PROPER BALANCE OF COMPETITIVE AND TECHNICAL RISKS

This is a basic fact of life which may not be fully appreciated by those not actively involved in industry. The total technology required to meet the requirement is indicated schematically by the area of the rectangular box. The total technology is composed of that which is available and that which must be demonstrated. The latter represents technical risk. If a design is proposed that rests solely on available technology, it runs the risk of being noncompetitive. On the other hand, a design which pushes too far beyond available technology runs a high technical risk. The means by which this critical balance is achieved are what tend to distinguish one organization from another. Schedule and funding constraints as well as the implications of falling short of the requirements obviously play big roles in the approach taken.

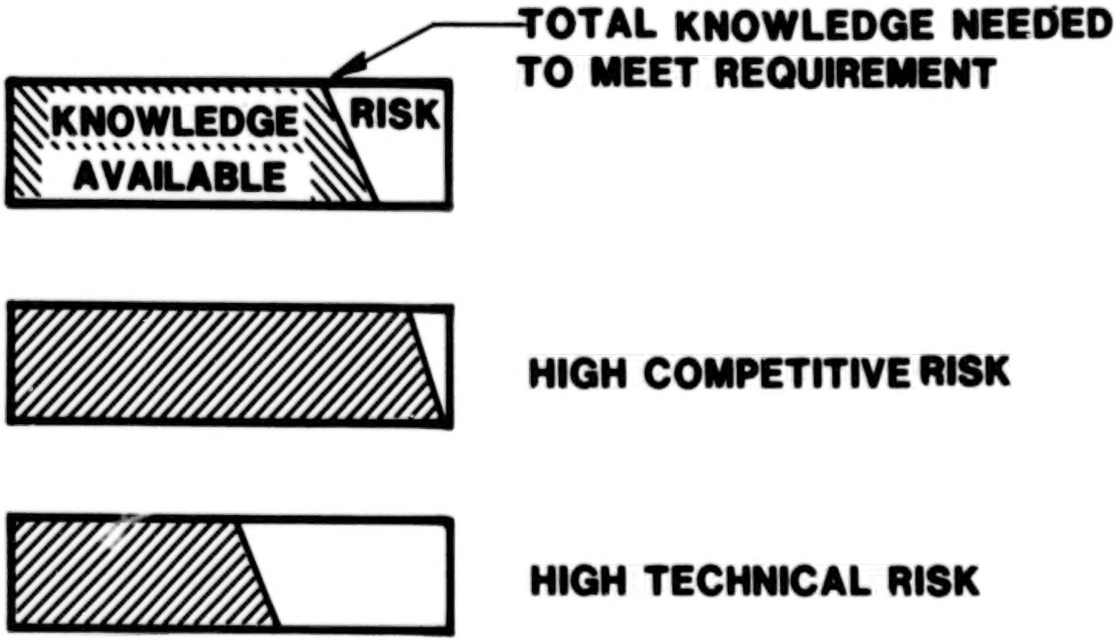
TOTAL KNOWLEDGE NEEDED
TO MEET REQUIREMENT
KNOWLEDGE
AVAILABLE
RISK
HIGH COMPETITIVE RISK
HIGH TECHNICAL RISK

Figure 5

ANALYSES TEND TO LAG HARDWARE AND TEST CAPABILITIES

It appears inevitable that availability of first principle analyses to reduce the risk in the design process will always be less than desired. This is due to several factors. The problem being dealt with is complex. Also designers seem to be able to invent new configurations/concepts faster than the analysts can develop their analyses. The figure below indicates the degree to which the development of advanced numerical rotor aerodynamic analyses have lagged the ability to build helicopters.

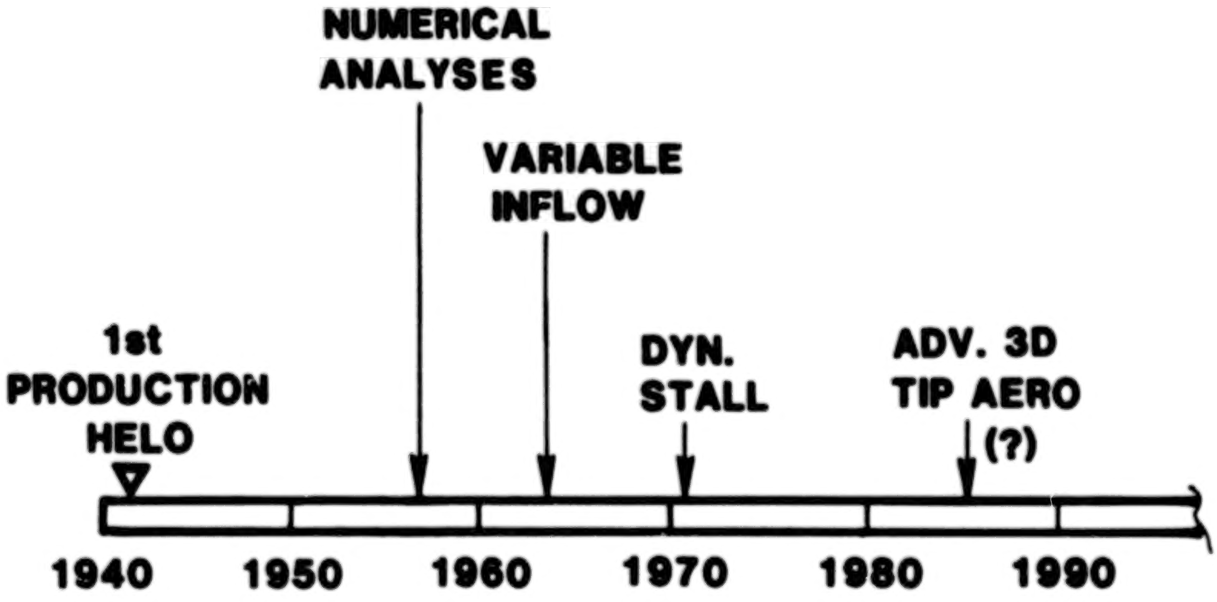

Figure 6

DESIGN ANALYSES OFTEN HAMPERED BY INPUT LIMITATIONS

It must also be recognized that even if an analysis has been developed, its usefulness may be critically hampered by the accuracy of the input data to the analysis. Detailed analyses often require detailed knowledge of the helicopter. Much of this knowledge only becomes available after the full system is available. There appears to be a consensus that given the measured detailed aerodynamic loading on a rotor, its noise could be predicted in detail. However, this is not of great help in the design phase.

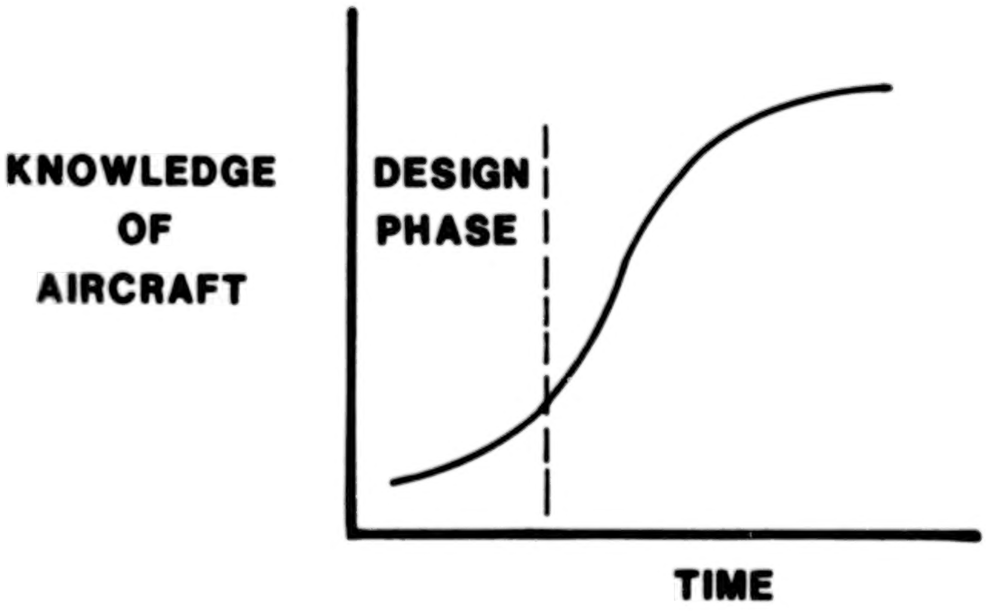

Figure 7

SUCCESSFUL ACOUSTIC PROGRAM REQUIREMENTS

Given this general background it is not surprising that the manpower loading profile of a typical new helicopter program looks as shown schematically below. Development problems will occur. The object is to contain them and to be able to solve them. What is needed is a proper blend of design tools, risk reduction tools and problem solving tools. Inadequacies in any one area may be compensated by strengths in another area. Thus acoustic design tools become less critical if there are "knobs to turn" to solve development problems. However, lacking knobs to turn, the design and risk reduction tools become extremely important.

NEEDED:

- **DESIGN TOOLS**
- **RISK REDUCTION TOOLS**
- **PROBLEM SOLVING TOOLS**

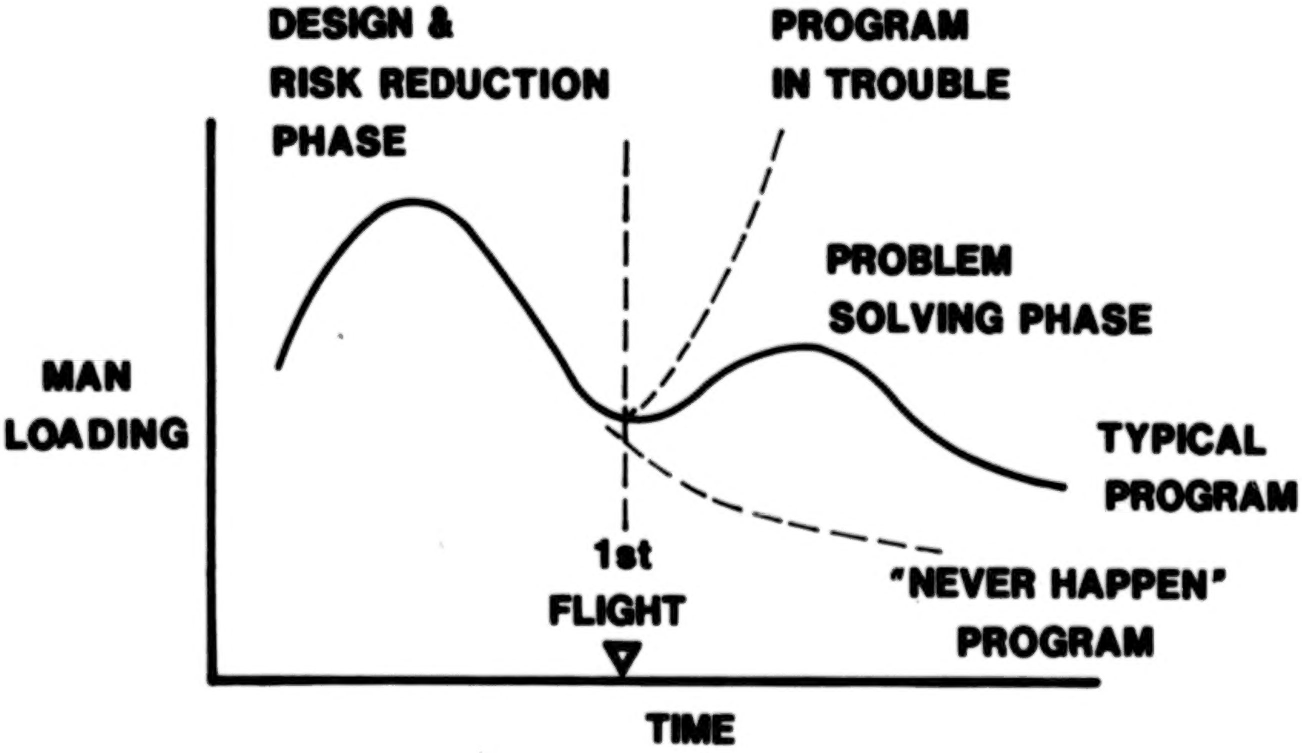
DESIGN & RISK REDUCTION PHASE
PROGRAM IN TROUBLE
PROBLEM SOLVING PHASE
MAN LOADING
TYPICAL PROGRAM
1st FLIGHT
"NEVER HAPPEN" PROGRAM
TIME

Figure 8

WHAT IS THE ACOUSTIC DESIGN REQUIREMENT?

There are several types of potential requirements facing the designer. Detectability is primarily a military issue in which main rotor harmonic noise is critical. ICAO and proposed FAA certification requirements are similar and involve limiting EPNL levels at three rigidly prescribed flight conditions: takeoff, flyover and landing. Higher frequency components of noise (broadband?) are more important in this situation. Local heliport regulations generally involve dBA noise generated during landing and takeoff. Flight path flexibility (within safe operation) is allowed in this situation. Depending on the limits selected, any of these requirements can drive the acoustic design of the helicopter. The remainder of the presentation focuses on the civil certification requirement.

- **DETECTIBILITY**
- **FAA CERTIFICATION (TEMPORARILY WITHDRAWN)**
- **ICAO CERTIFICATION**
    - **TAKEOFF, FLYOVER, LANDING EPNL**
- **LOCAL HELIPORT**

Figure 9

HOW SEVERE IS THE POTENTIAL REQUIREMENT?

The figure below compares the noise limit in flyover that had been proposed by the FAA (and now adopted by ICAO) with available preliminary data on the helicopters that would be in commercial production in 1985. The vertical lines shown represent the estimated data accuracy (±2 EPNdB). It is clear that the limit does lie close to the noise levels generated by the current and near future fleet of aircraft. Many of these helicopters include noise reduction features such as moderate tip speeds, disc loadings, improved thinner airfoils and advanced tips. Future helicopters will require further technology improvements if they are to be designed with confidence in meeting such a certification requirement without significantly impacting other important attributes of the aircraft such as cruise speed, gross weight and operating cost.

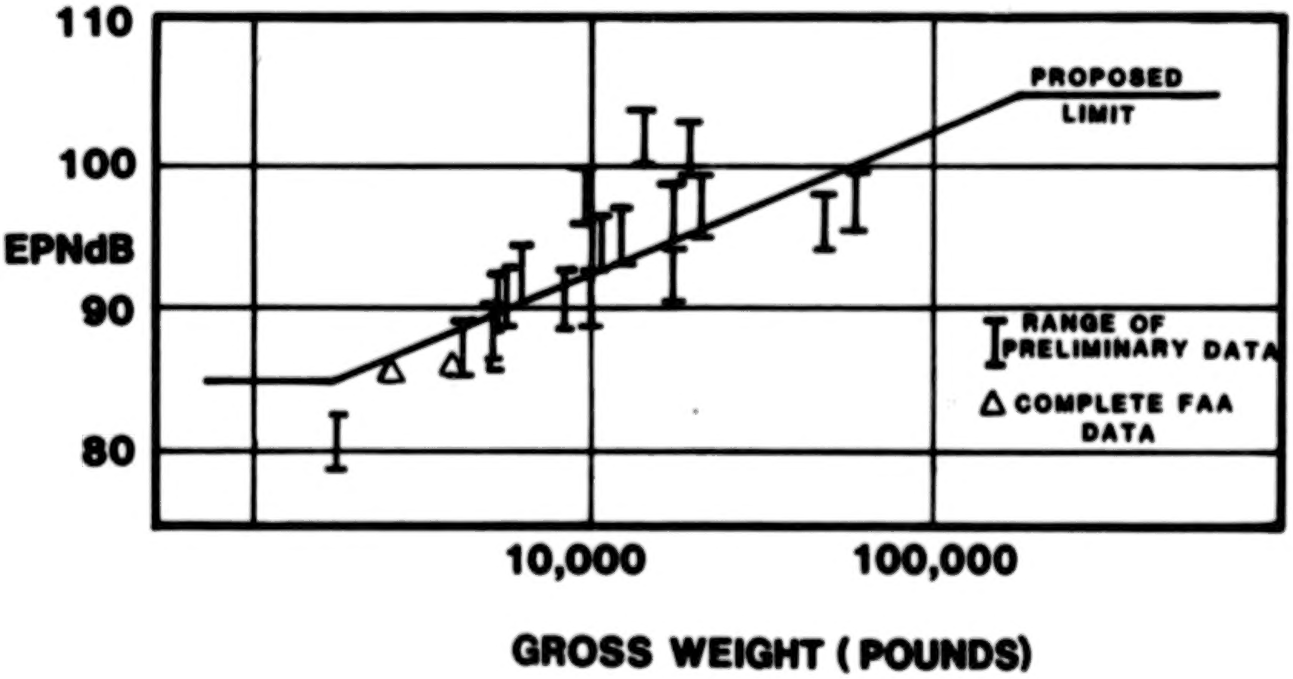

Figure 10

SOME DETAILS OF THE CERTIFICATION ACOUSTIC PROBLEM

EPNL is the metric employed in the proposed certification rules. There are several steps in the computation of EPNL. These are illustrated below. Because of the weightings involved, the principal contributors to EPNL may not always be obvious at the time history level. For this reason it is imperative that a bottom-line "complete" prediction analysis be available so that the real accuracy of any proposed method can be established and so that design trades can be performed.

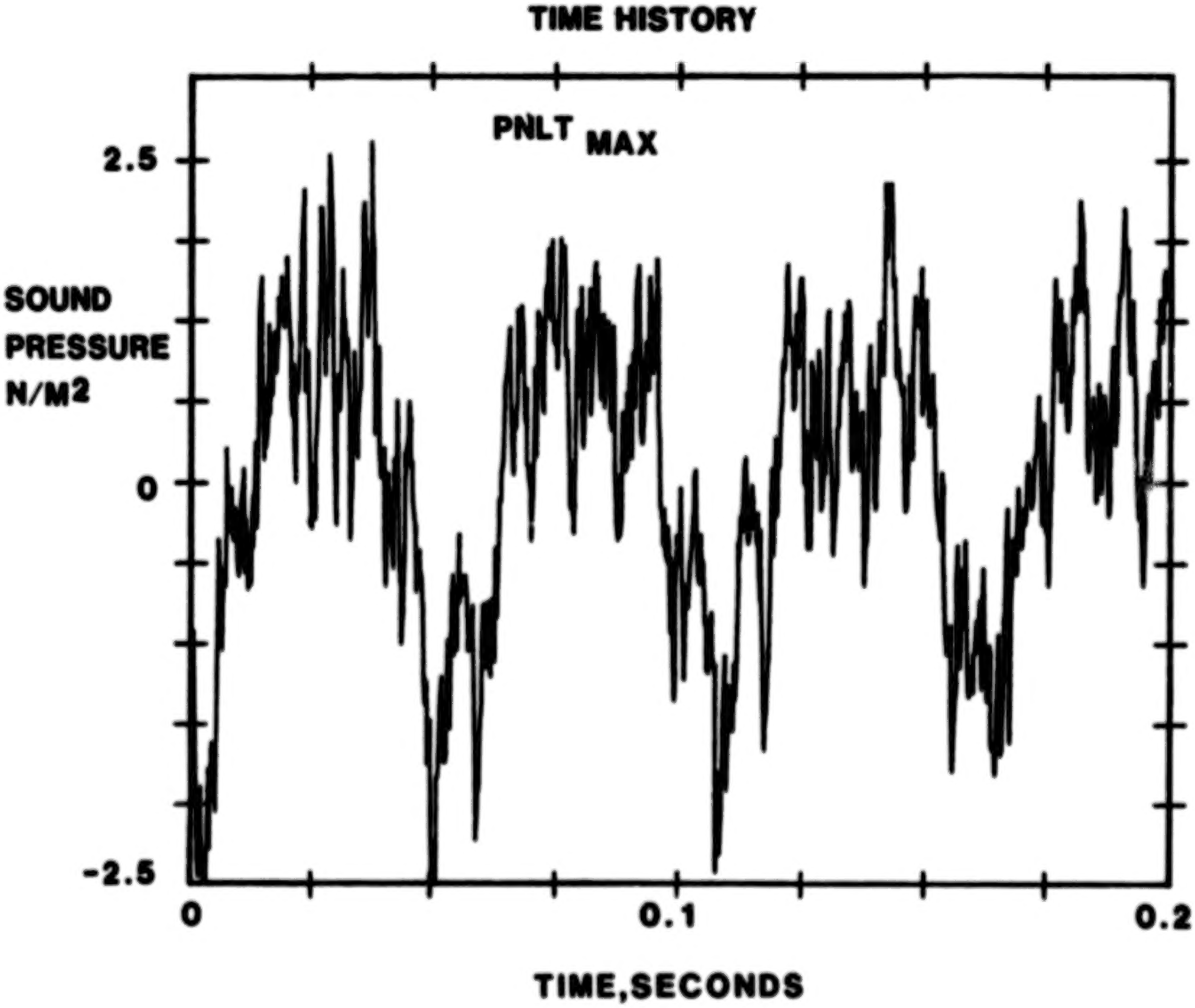

Figure 11

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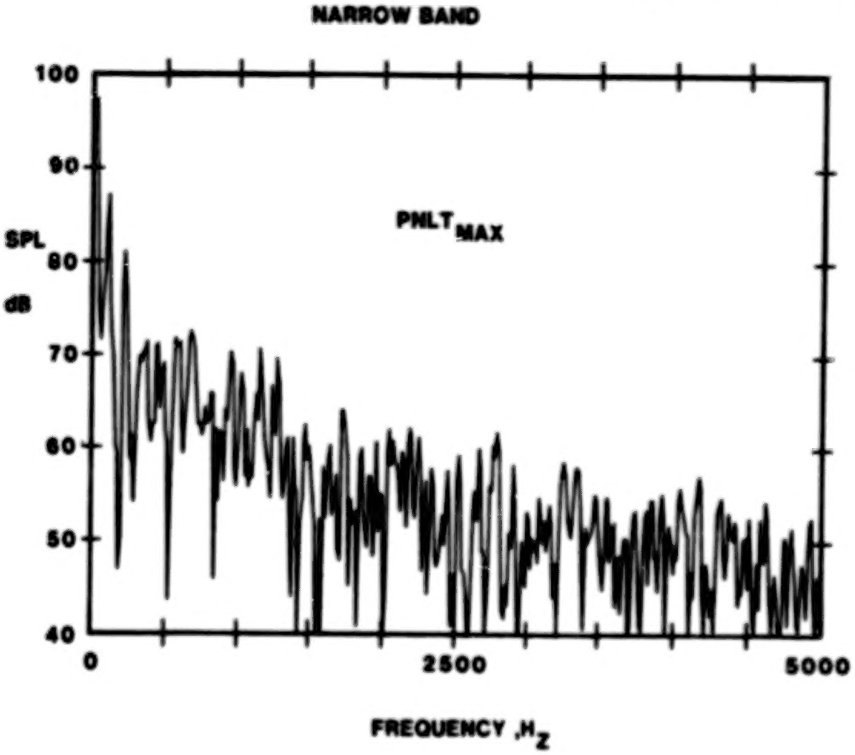

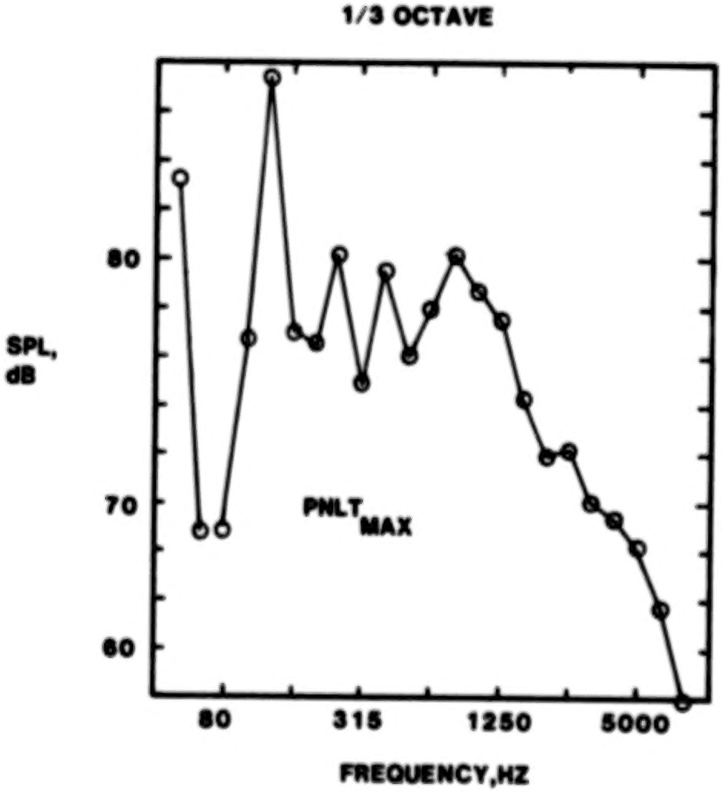

Figure 11 (continued)

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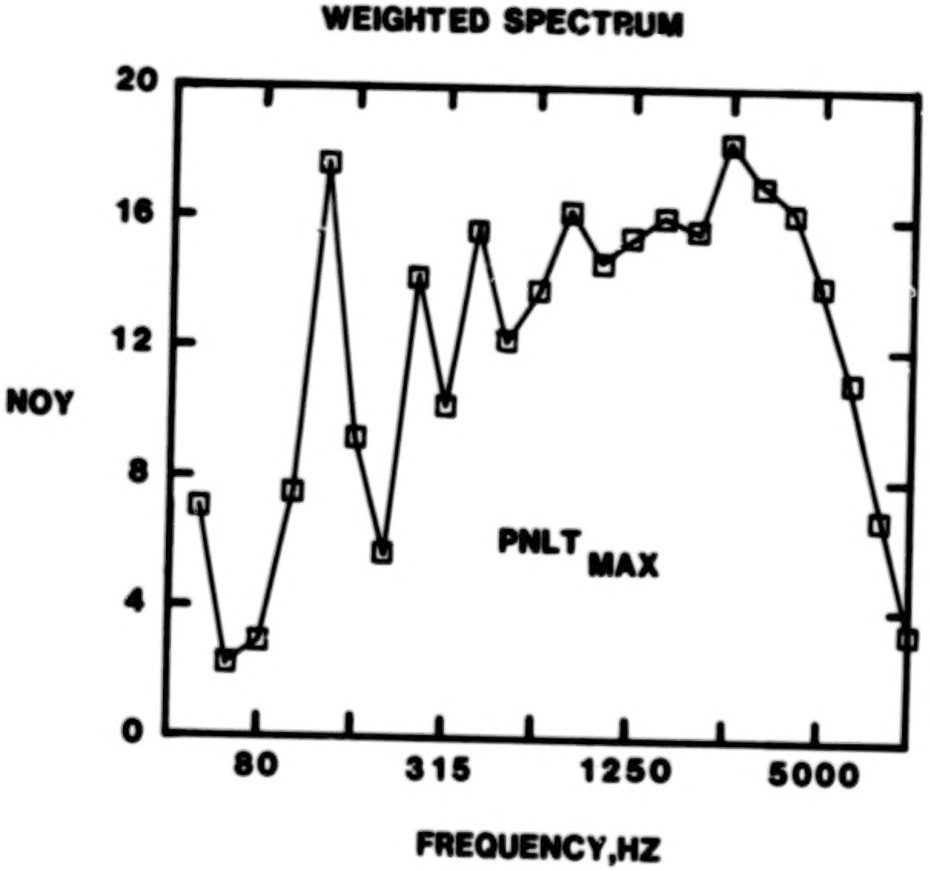


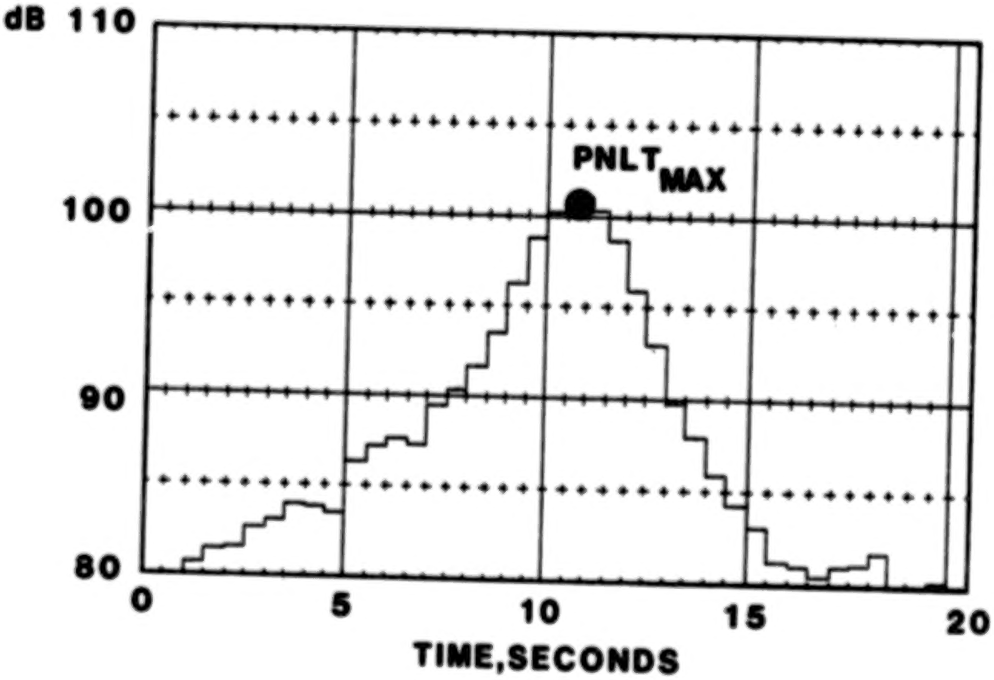


Figure 11 (concluded)

ACOUSTIC DESIGN AND RISK REDUCTION TOOLS

What are the design and risk reduction tools available to the acoustician/designer? Exact organization of these tools can be the subject of debate. However, for the purposes of this paper, the tools have been divided into three major categories. First-principle analyses are defined as those which nominally only require as input the physical and operational parameters of the helicopter in order to compute its acoustic characteristics. Semi-empirical analyses combine a data base of reasonable scope with some analytic guidance to allow the data base to be generalized so that predictions for a new design can be made. Empirical analyses are defined as those using data more or less directly for design. Most helicopter acoustic analyses tend to be of the semi-empirical type.

FIRST PRINCIPLE ANALYSES

- MECHANISM VS COMPONENT VS SYSTEM
- ABSOLUTE VS TRENDING

SEMI-EMPIRICAL ANALYSES

- SYSTEM
- COMPONENT
- MECHANISM

EMPIRICAL ANALYSES

- MODEL
- FLIGHT TEST

Figure 12

REQUIREMENTS FOR SYSTEM FIRST PRINCIPLE ANALYSIS

Obviously, first principle analyses are very desirable since the greatest understanding and least risk result. One of the problems facing all of us involved in helicopter acoustics is to define all of the elements likely to be required in a first principle analysis. The analysis features will vary depending on the acoustic requirement which the helicopter is being asked to meet. Thus a first principle analysis for predicting helicopter detectability characteristics might be quite different from that required for predicting EPNL certification levels. Listed below are the elements of such an EPNL analysis as viewed by Sikorsky. Having defined the list of requirements, it remains to develop an analysis fulfilling (to some clearly defined degree) those requirements and to evaluate its accuracy in predicting the "bottom line" attribute that the designer is concerned with. It is our perception that although progress is being made on isolated fronts, a first principle analysis simply is not available at the present time and probably will not be available in the near future.

1. ROTOR NOISE
    - 1.1 THICKNESS
    - 1.2 LEADING EDGE
        - TURBULENCE
        - FUSELAGE
        - SELF-INDUCTION
        - TR ON MR
        - MR ON TR
        - SURFACE PRESSURE
        - BLADE RESPONSE
    - 1.3 TRAILING EDGE
        - BOUNDARY LAYER
    - 1.4 QUADRUPOLE
        - OFF BODY FLOW
    - 1.5 TIP VORTEX FORMATION
    - 1.6 WAKE

2. FUSELAGE NOISE

3. ENGINE NOISE
    - 3.1 COMPRESSOR
    - 3.2 CORE
    - 3.3 EXHAUST

4. SPECIAL FEATURES
    - 4.1 ARBITRARY ORIENTATION & ROTATION OF ROTORS
    - 4.2 MULTIPLE ROTORS

5. ACOUSTIC PROPAGATION
    - 5.1 ATMOSPHERIC EFFECTS
    - 5.2 NONLINEAR

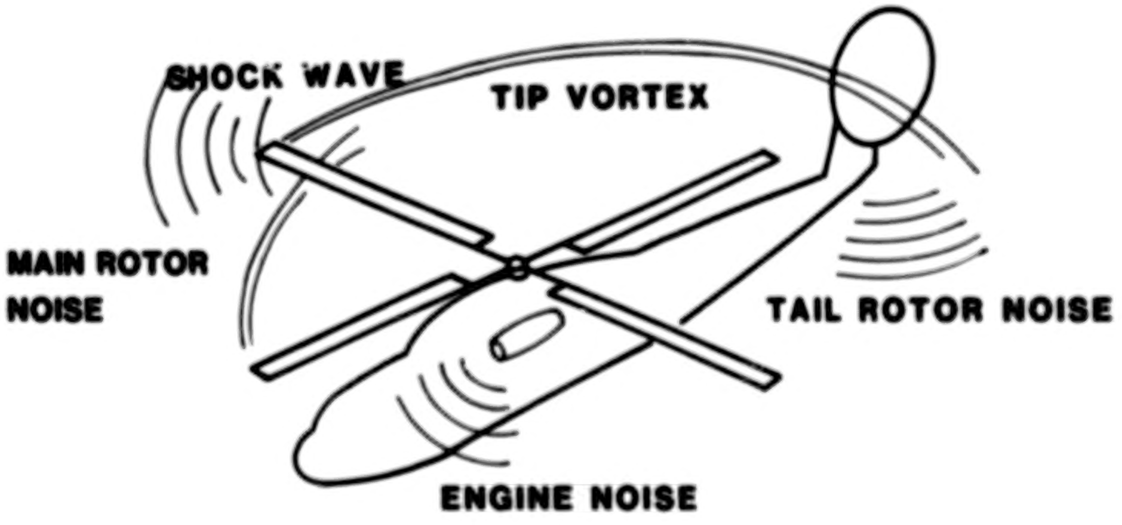
6. MEASUREMENTS AT OBSERVER
    - 6.1 GROUND REFLECTION

7. USER-ORIENTED FEATURES
    - 7.0 PROGRAM AVAILABILITY
    - 7.1 RUNNING TIME
    - 7.2 COUPLING OF AERO AND ACOUSTIC PROGRAMS
    - 7.3 ARBITRARY FLIGHT PATH
    - 7.4 ARBITRARY OBSERVER POSITION
    - 7.5 PERTINENT OUTPUT
    - 7.6 USER'S MANUAL
    - 7.7 DOCUMENTATION
    - 7.8 PERTINENT CHECK CASES
    - 7.9 PERTINENT VALIDATION CASES

Figure 13

EMPIRICAL DESIGN REQUIRES A SUBSTANTIAL DATA BASE

To approach a new design from a purely empirical approach requires a substantial, largely-existing data base. The number of design and operating parameters that can influence the acoustic signature is large, as shown below, and by no means does systematic data exist to define the effects of each parameter. As a practical matter, then, this option is not now available to the acoustic engineer, particularly for new designs. For derivatives of existing aircraft the empirical approach becomes more practical.



PARAMETERS OF INTEREST

- **TIP SPEED**
- **FWD SPEED**
- **GROSS WEIGHT/DRAG**
- **POWER**
- **RATE OF CLIMB/DESCENT**
- **TWIST**
- **AIRFOIL**
- **NO. OF BLADES**
- **TIP SHAPE**
- **TAIL ROTOR -PYLON VARIABLES**
- **MAIN ROTOR WAKE -TAIL ROTOR**
- **ENGINE INLET/CORE/ EXHAUST**
- **AMBIENT CONDITIONS**

Figure 14

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SEMI-EMPIRICAL ANALYSES FORM THE BASIS OF DESIGN

Lacking a first principle analysis, acoustic design proceeds using semi-empirical analyses. Two types of analyses are available at Sikorsky. The first applies at the complete system level. Here system EPNL data are plotted against a generalized parameter derived from the Ffowcs-Williams formulation of the acoustic loading term. In the second analysis the EPNL is computed from the main rotor, tail rotor and engine component contributions. The empiricism here involves the prescription of the roll-off factor for main and tail rotor harmonics of loading, the constants for broadband noise and engine noise characteristics.

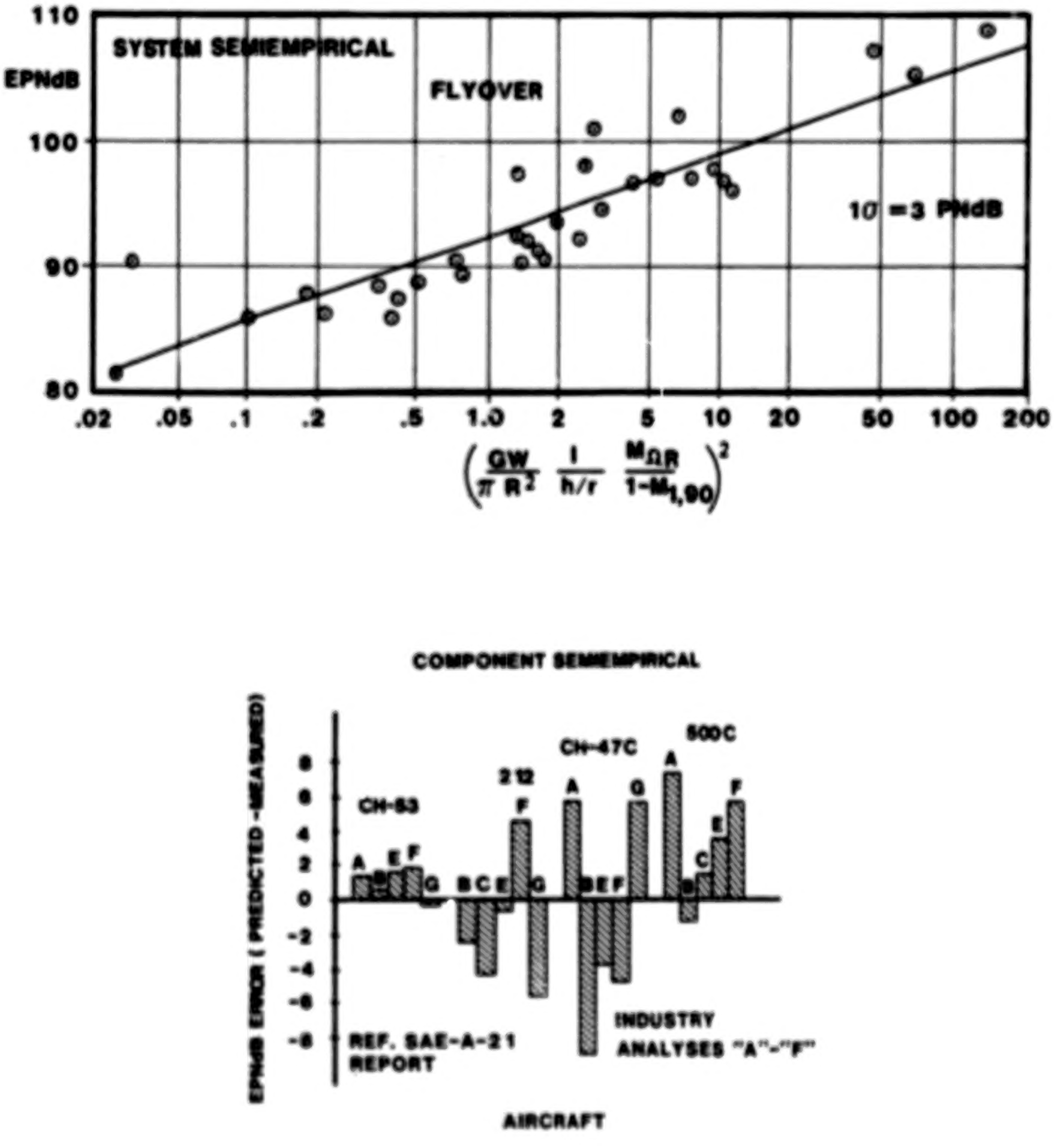

Figure 15

RISK REDUCTION OPTIONS

As discussed earlier, risk reduction efforts play a vital role in the overall design/development process and the need for acoustic risk reduction efforts must be anticipated. Because of the severe pressures of schedule, it is imperative that risk reduction techniques be both responsive and previously validated prior to their use. Possible acoustic risk reduction approaches are noted below. They basically involve trying to simulate the proposed design and/or operating conditions as closely as possible through model and/or full scale flight test. The use of model wind tunnel tests in this role has yet to be validated - particularly for application to the EPNL moving source problem. Flight tests on the other hand, may not be able to simulate the proposed design as closely as desired. The net result is that probable conservatism will be forced into the design.

- **WIND TUNNEL TESTING (MODEL)**
- **FLIGHT TESTING**

Figure 16

PROBLEM SOLVING REQUIREMENTS

Problem solving also represents a vital part of design/development process. Problems do arise regardless of the best efforts in the design and risk reduction phases. Acoustic problems must likewise be anticipated. When they do, the engineer ideally should have practical "knobs to turn" that will contain the problem to acceptable levels. As noted in the figure below, the number of high confidence knobs available to the acoustics engineer is not very large, and more importantly they tend to be very unpalatable. As a result, conservatism is again forced into the original design.

PROGRAM MANAGER:

IT MAKES TOO MUCH NOISE!!

ACOUSTICS ENGINEER :

WELL, HOW ABOUT REDUCING

RPM OR GROSS WEIGHT OR

CRUISE SPEED BY 25% ?

Figure 17

PROCEDURE RECENTLY RECOMMENDED FOR NEW DESIGN AIRCRAFT

By now it should be evident that the design/development procedure is by no means cut and dry but rather involves proper blending of design, risk reduction and problem solving skills. In a recent design study for a new aircraft the following recommendations were made and are indicative of how capabilities that existed at the time were judged.

- **ACCOUNT FOR BOTH TEST AND ANALYSIS UNCERTAINTIES**
- **DESIGN FOR SELECTED CONFIDENCE OF PASSING CERTIFICATION**
- **ALLOW MARGIN FOR DERIVATIVE AIRCRAFT**
- **USE SEMI-EMPIRICAL SYSTEM ANALYSIS**
- **INCORPORATE ALL DESIGN FEATURES EXPECTED TO HAVE A FAVORABLE ACOUSTIC EFFECT**
- **BUILD IN PROVISIONS FOR CHANGING TIP DESIGN OF MAIN ROTOR**
- **SIMULATE THE DESIGN WITH FLIGHT TEST OF THE CLOSEST POSSIBLE EXISTING AIRCRAFT AND ACQUIRE DESIGN TRADE INFO**
- **CONDUCT AN ACOUSTIC WIND TUNNEL TEST OF A MAIN ROTOR/ TAIL ROTOR MODEL**

Figure 18

IMPACT OF ACOUSTIC DESIGN REQUIREMENT ON NEW DESIGN AIRCRAFT

The figure below shows the impact of the proposed FAA acoustic regulation on a possible new civil helicopter design. The design procedure referred to in Figure 18 was employed. The results show significant increases in both gross weight and direct operating costs as the probability of passing the regulation being proposed by the FAA at the time was increased and as the margin allowed for derivative aircraft was increased. Note that penalties are much smaller at the 50% probability level. This is not surprising since as shown in Figure 10, the regulation levels as proposed were about equal to the noise levels generated by the current state of the art.

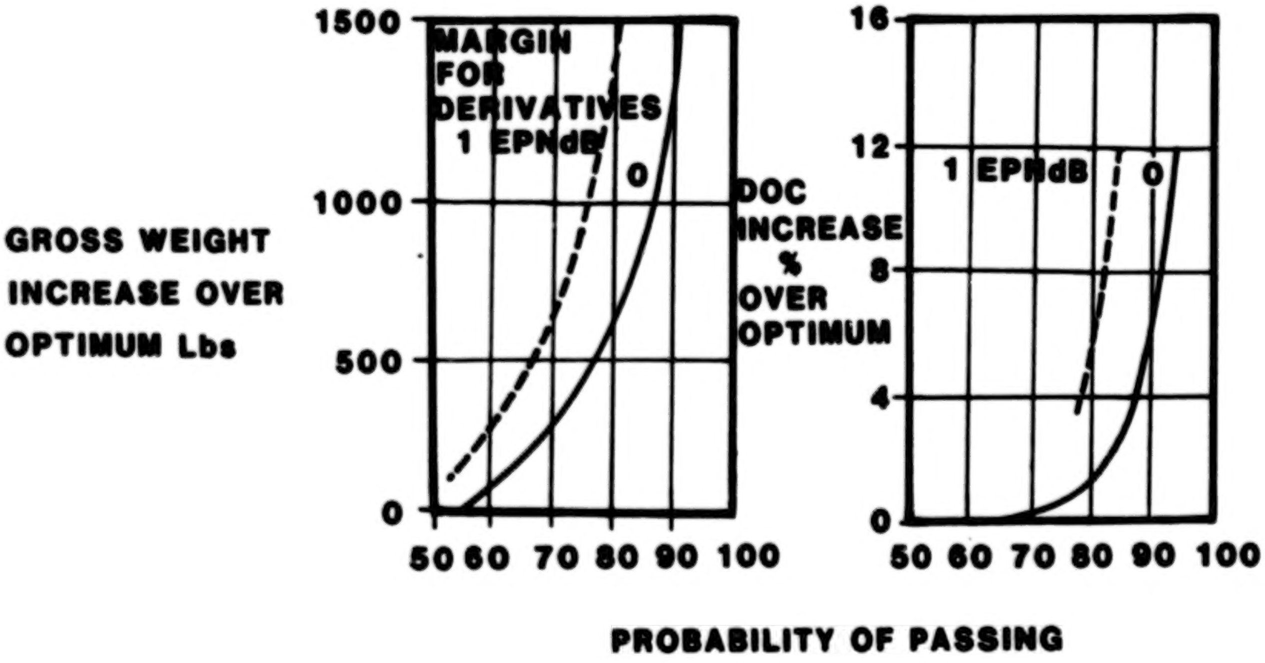

Figure 19

REQUIREMENTS FOR DERIVATIVE AIRCRAFT

The figure below shows the two basic situations that can exist for derivative aircraft relative to the regulation originally proposed by the FAA. In situation 1 the parent aircraft noise level is less than the requirement. If in this situation the weight of the aircraft is increased for a derivative version, the noise is allowed to grow to that of point B. If, in situation 2 the noise level of the parent aircraft is above the requirement (this is possible with the tradeoff provisions that were proposed) no increase in noise would be allowed for the derivative. In the case where speed increases at a fixed gross weight were planned for the derivative, the same constraints would apply except in this case the weight at points A and B would be the same.

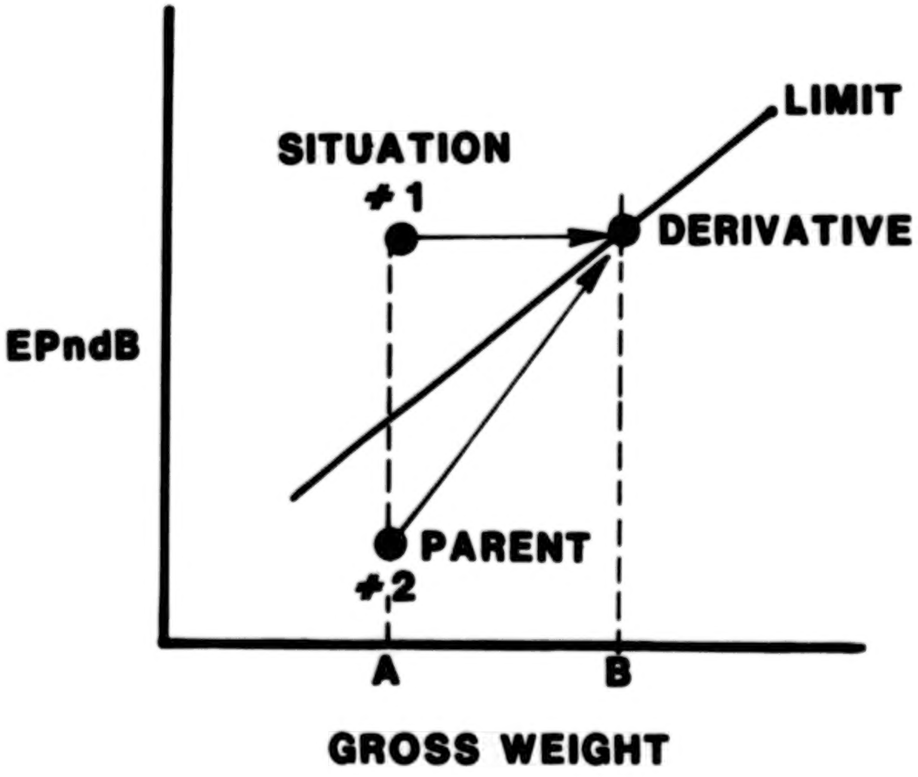

Figure 20

PROCEDURE RECOMMENDED FOR DERIVATIVE AIRCRAFT

The design procedure recommended for a derivative aircraft is largely an empirical one made possible by the existence of the parent aircraft. The procedure is straightforward as long as the aircraft rotor characteristics are not being altered. Changes in operational parameters such as gross weight and speed can be evaluated directly by test. The accuracy of the test is a concern as is the ability to repeat the result during any official certification test.

- **TEST PARENT AIRCRAFT AT CONDITIONS SIMULATING THE DERIVATIVE AIRCRAFT**
- **ESTIMATE DELTAS DUE TO ROTOR DESIGN CHANGES USING CORRELATED COMPONENT SEMI-EMPIRICAL ANALYSIS**
- **ACCOUNT FOR ANY CHANGES IN ENGINE CONTRIBUTION USING ENGINE MANUFACTURE STATIC NOISE DATA**
- **DEFINE ACCURACY OF TEST AND ANALYTIC RESULTS**
- **PROJECT NOISE LEVELS TO SELECTED LEVEL OF CONFIDENCE**

Figure 21

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IMPACT OF ACOUSTIC REQUIREMENT ON DERIVATIVE AIRCRAFT

Currently, the question of how to treat derivative aircraft under the ICAO rule is under discussion. The FAA rule that was originally proposed presented a severe problem for derivative aircraft. This was due primarily to (a) the limit levels that were comparable to parent aircraft levels, (b) an allowed growth of noise with gross weight that was too low for a fixed diameter rotor situation (see figure below), and (c) no allowance in the rule for growth in noise due to higher forward speeds. As a result, for many aircraft the normal derivative process involving growth in productivity (due to weight and speed improvements) with fixed rotors would be prevented if such a rule were adopted.

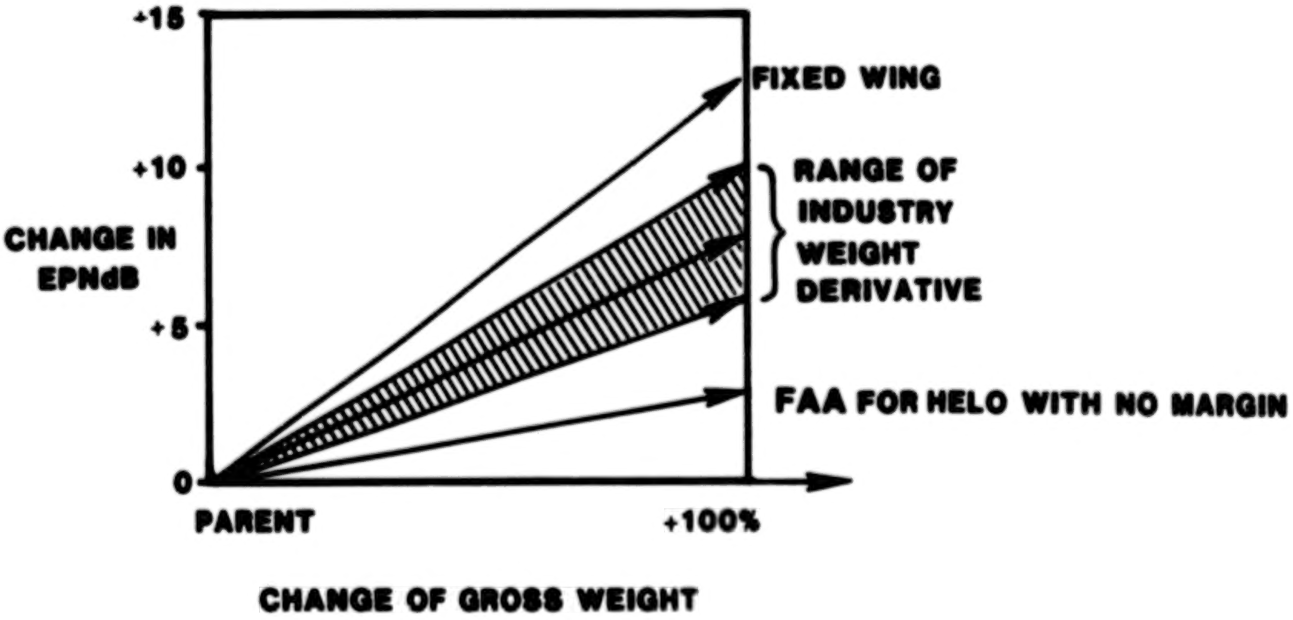


Figure 22

EXPERIENCE IN REDUCING NOISE

Sikorsky helicopters historically have been relatively quiet. This has resulted primarily from the multi-bladed, single-rotor configuration that has allowed both moderate tip speeds to be used and certain inter-rotor blade vortex interactions to be avoided. Main rotor wake interaction with the tail rotor, of course, is still present and is of concern. However, in general the external noise of our machines has been a relatively contained problem. As a result, experience with specific efforts to reduce noise is relatively limited. One investigation of interest was the HUSH Program in which a relatively unconstrained effort to reduce the noise of an S-61 helicopter was conducted. The effort involved silencing engines and slowing down the tail rotor and the main rotor. The principal changes to the system and results obtained are noted below. As noted in the figure, noise reductions were possible but with the technology in hand at the time a significant penalty in gross weight resulted.

	STD	HUSH
MAIN ROTOR		
TIP SPEED	**100%**	**90%**
# OF BLADES	**5**	**6**
TIPS	**SQUARE**	**TRAPEZOIDAL**
TAIL ROTOR		
TIP SPEED	**100%**	**66%**
# OF BLADES	**5**	**10**
ENGINE SILENCING		
INLET	**NO**	**YES**
EXHAUST	**NO**	**YES**
VERTICAL TAIL	**UNCAMBERED**	**CAMBERED**
NOISE LEVEL EPNL	**REFERENCE**	**REF- 2-4dB**
DETECTABILITY DISTANCE	**100%**	**65%**
WEIGHT INCREASE	**-**	**6.6% DGW**

Figure 23

EFFECT OF NEW DESIGN FEATURES IN S-76 AND BLACK HAWK

In developing our more recent aircraft, the noise attribute was monitored and design features incorporated for other reasons which were believed to have beneficial effects on noise were strongly supported. As a result, our current generation of aircraft generates about 2 EPNdB less flyover noise for a given nondimensional design condition than does the previous generation of aircraft. This is shown in the figure below. The lower noise is attributed to the use of thinner advanced airfoils and swept/tapered tips. It should be noted, however, that the noise benefit implied by these results has not always been realized since other mission requirements have led to increased values of forward speed, tip speed and disc loading for the same gross weight.

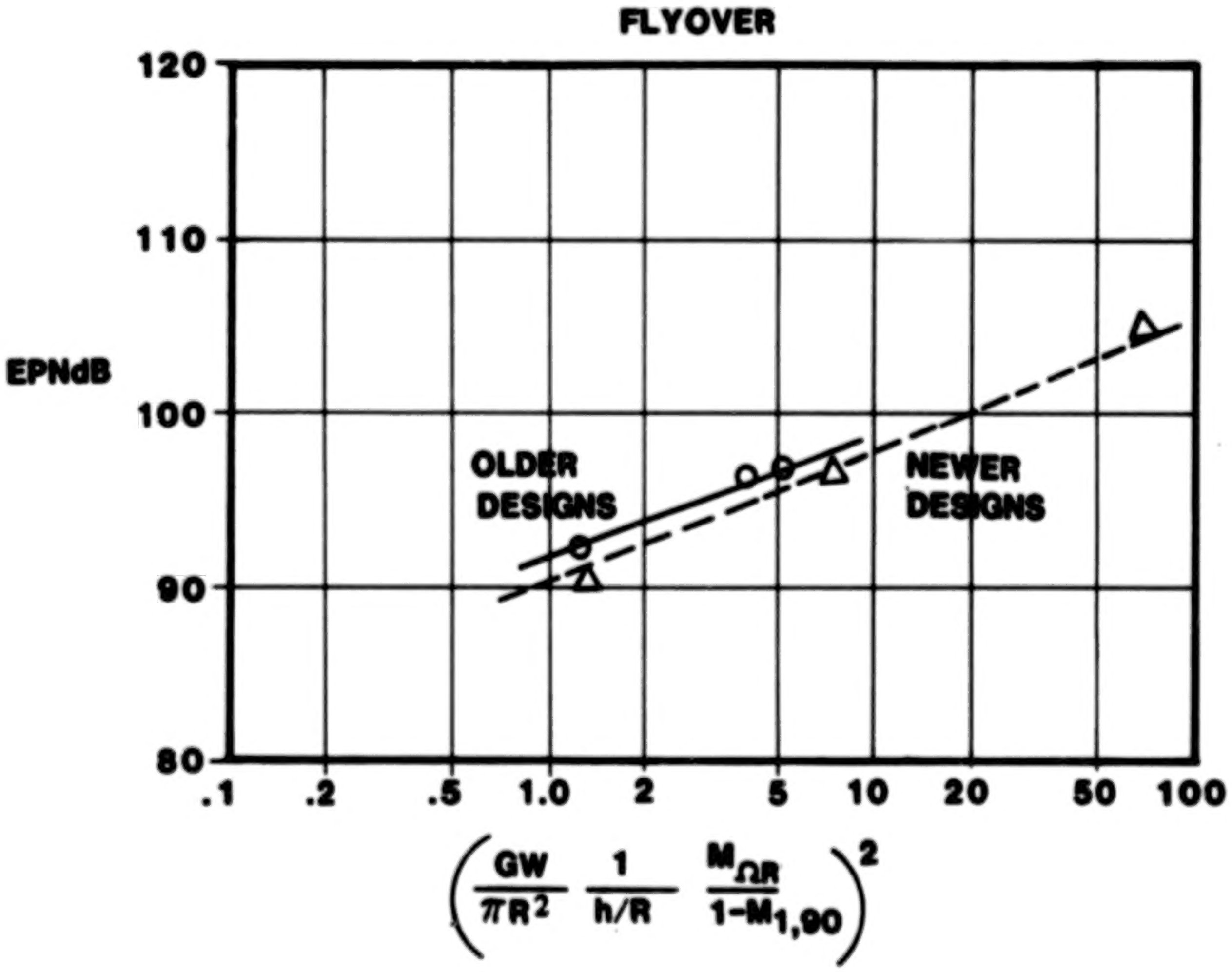


Figure 24

CONCLUSIONS

The conclusions which I believe are valid based on the material presented are listed below.

- THE TOOLS CURRENTLY AVAILABLE FOR ACOUSTIC DESIGN RISK REDUCTION AND PROBLEM SOLVING ARE LIMITED IN ACCURACY AND DO NOT ENCOMPASS ALL DESIGN PARAMETERS.
- ASSUMING AVAILABLE TECHNOLOGY ,DESIGNING HELICOPTERS TO CONFIDENTLY MEET NOISE LIMITS AS WERE PROPOSED BY THE FAA AND HAVE NOW BEEN ADOPTED BY ICAO WILL RESULT IN SIGNIFICANT DESIGN COMPROMISES.

Figure 25

N84 29664

NOISE REDUCTION TECHNIQUES -
CRITERIA AND VALIDATION

J. M. Drees
Bell Helicopter Textron
Fort Worth, Texas

Because this paper was not available at time of publication, only slides are presented.

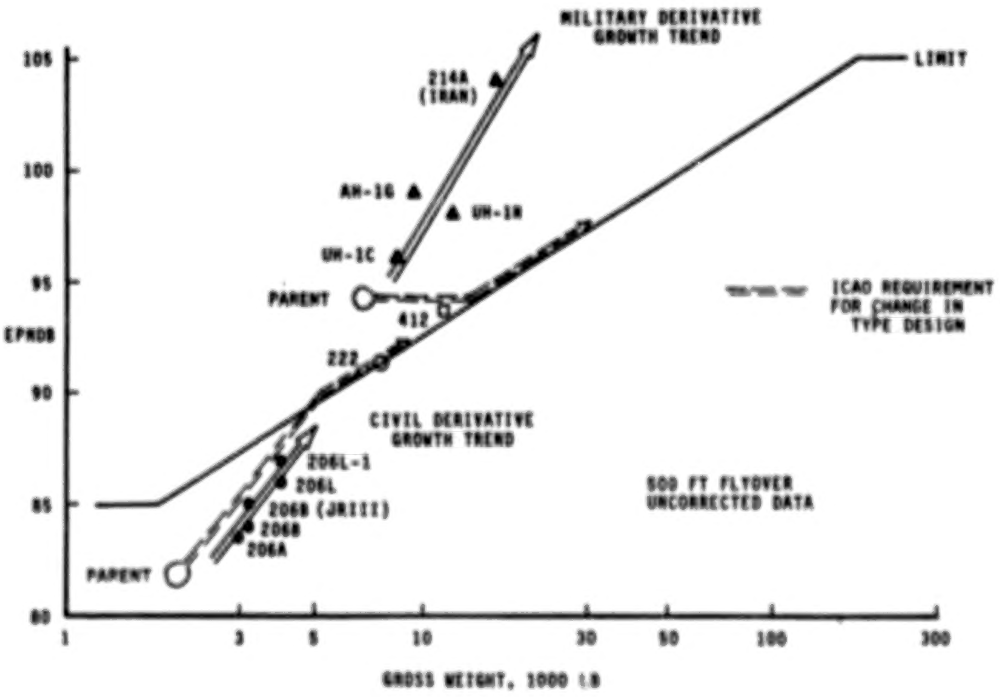
HISTORICAL OVERVIEW

PRIOR TO 1970
- NO CONSIDERATION TO NOISE REDUCTION
- DESIGN FOR HIGHEST EFFICIENCY

1970 - 1980
- FAILED ATTEMPTS TO USE FIRST PRINCIPLES
- WIND TUNNEL TEST, SCHLIEREN
- EMPIRICAL METHODS: Δ METHODS
- INHOUSE NOISE CRITERIA
- MATCHING MR & TR NOISE
- TIPSPEED EFFECT: SPECIAL TIPS

1980 - NOW
- ICAO/FAA RULE MAKING EFFORTS RESULTING IN NEW DESIGN CRITERIA
- NEED MORE RESEARCH
- OPERATIONAL CRITERIA

DERIVATIVES



ROTOR WING ANALOGY TO FIXED WING

F/W	WING	EMPENNAGE	CONTROLS	✓ ENGINES ----	
R/W	✓ MAIN ROTOR ----	✓ TAIL ROTOR ----	CONTROLS ----	ENGINES	DRIVE SYSTEM ----

✓ = PRIMARY NOISE PRODUCERS

---- = MUST BE CHANGED TO REDUCE NOISE

FLYOVER

DESIGN EMPHASIS

ADVANCING BLADE TIP MACH NUMBER

BLADE TIP AIRFOIL AND SHAPE

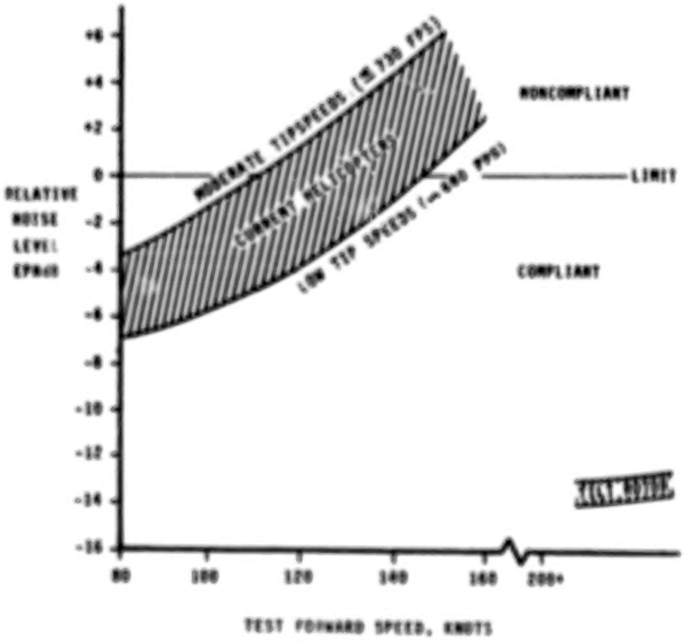
ICAO STANDARDS

CONDITIONS	INDIVIDUAL REQUIREMENTS
1. FLYOVER	• LESSER OF 0.9 V_{NE} OR 0.9 V_H • 150 METERS ALTITUDE
2. APPROACH	• 6° GLIDE SLOPE • V_Y AIRSPEED • 120 METERS ALTITUDE OVER MICS
3. TAKEOFF	• BEST RATE OF CLIMB AIRSPEED V_Y FROM 20 METERS ALTITUDE 500 METERS PRIOR TO MIC CROSSING

COMMON REQUIREMENTS

- MAXIMUM GROSS WEIGHT
- MAXIMUM NORMAL OPERATING RPM
- 6 PASSES FOR EACH CONDITION
- POSITION TRACKING
- METEOROLOGICAL MEASUREMENTS
- WIND LESS THAN 10 KNOTS
- 3 MICROPHONE ARRAY

AIRSPEED BARRIER



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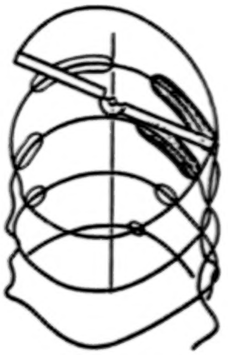
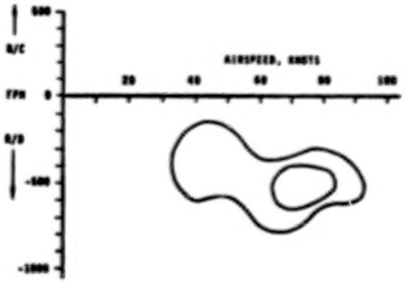
APPROACH



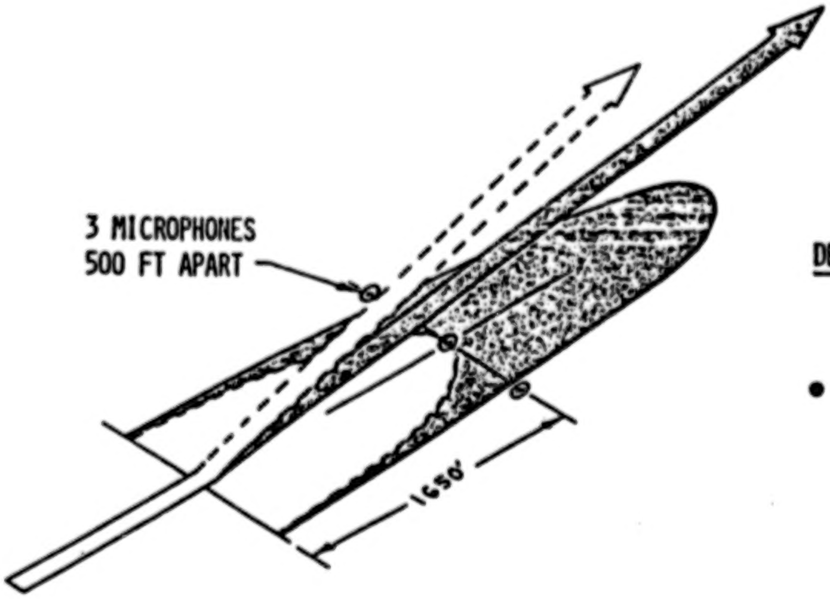
$\mu = 0.15$

DESIGN EMPHASIS

- AIRFOIL CHARACTERISTICS (FLUCTUATING LOADS)
- TIP VORTEX STRENGTH
- ADVANCE RATIO



TAKEOFF


3 MICROPHONES
500 FT APART
1650'

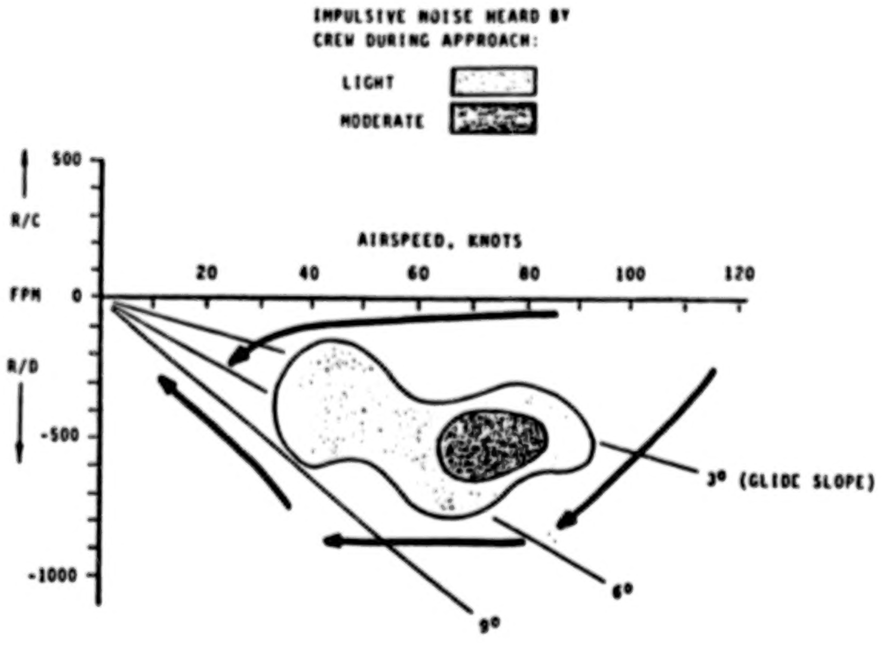
DESIGN EMPHASIS

- CLIMB PERFORMANCE

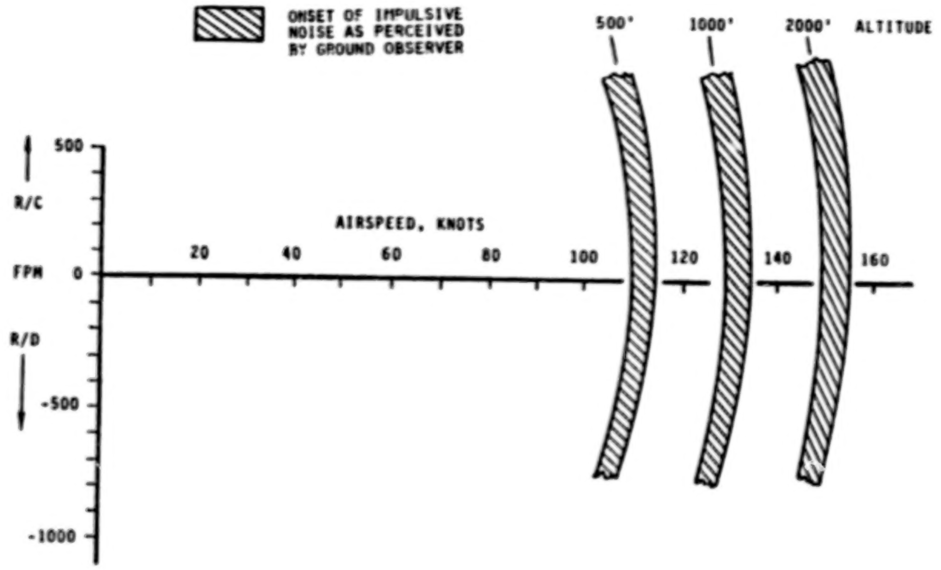
OPERATIONAL

- BLADE SLAP AVOIDANCE
- MINIMIZE FLY OVER NOISE
- DETECTION
- REDUCE TAKEOFF NOISE

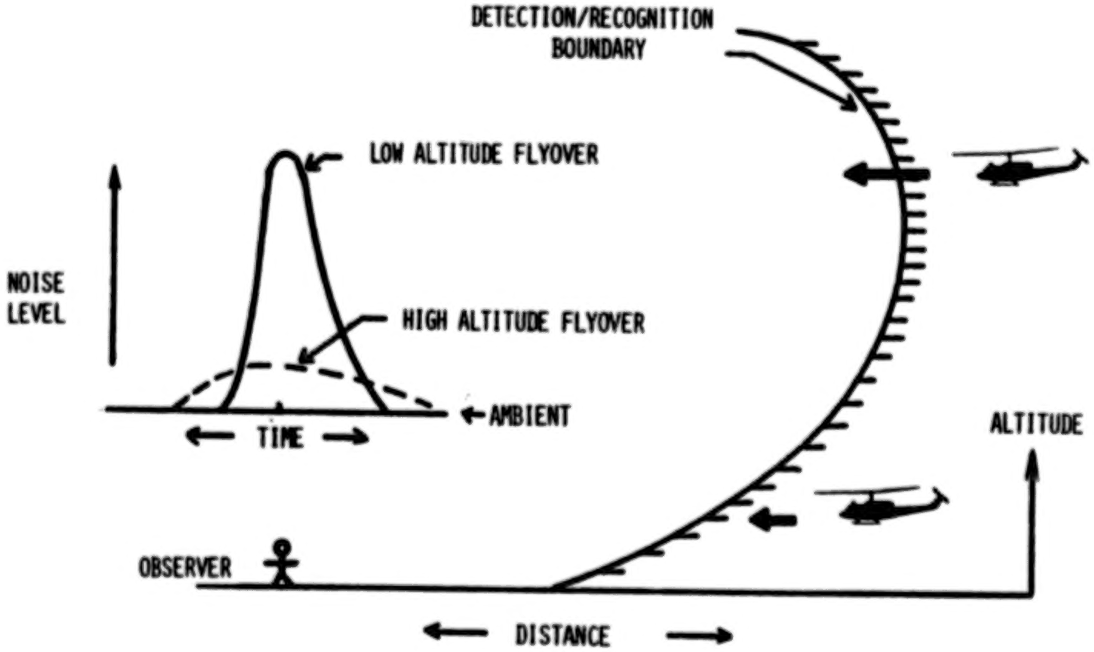
FLIGHT OPERATION CONSIDERATIONS



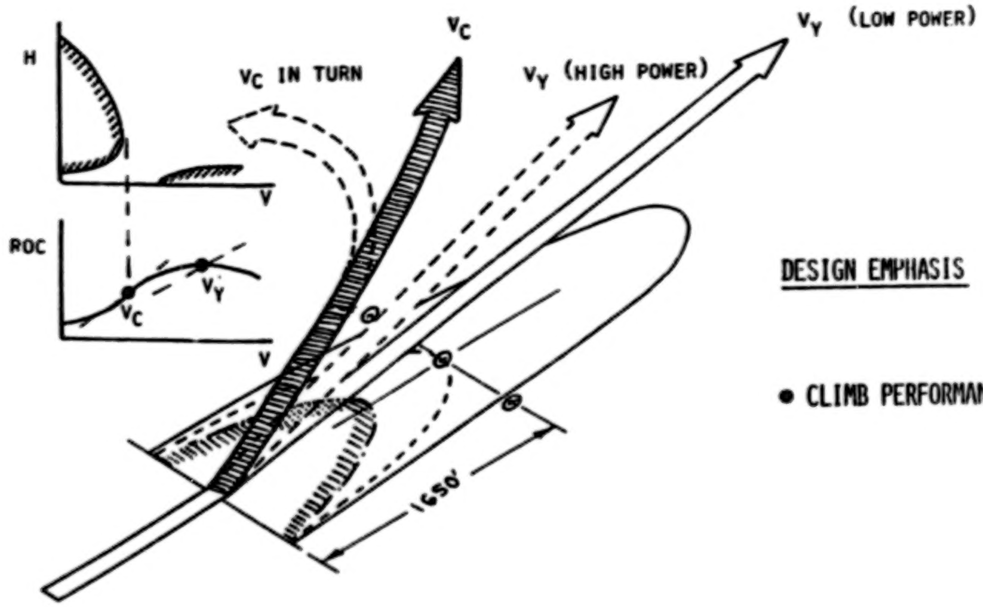
ALTITUDE/AIRSPEED EFFECTS



AURAL CHARACTERISTICS


DETECTION/RECOGNITION BOUNDARY
LOW ALTITUDE FLYOVER
NOISE LEVEL
HIGH ALTITUDE FLYOVER
AMBIENT
TIME
ALTITUDE
OBSERVER
DISTANCE

TAKEOFF AT V_C INSTEAD OF V_Y


H
V
ROC
V_C
V_Y
V
V_C IN TURN
V_C
V_Y (HIGH POWER)
V_Y (LOW POWER)
1650'

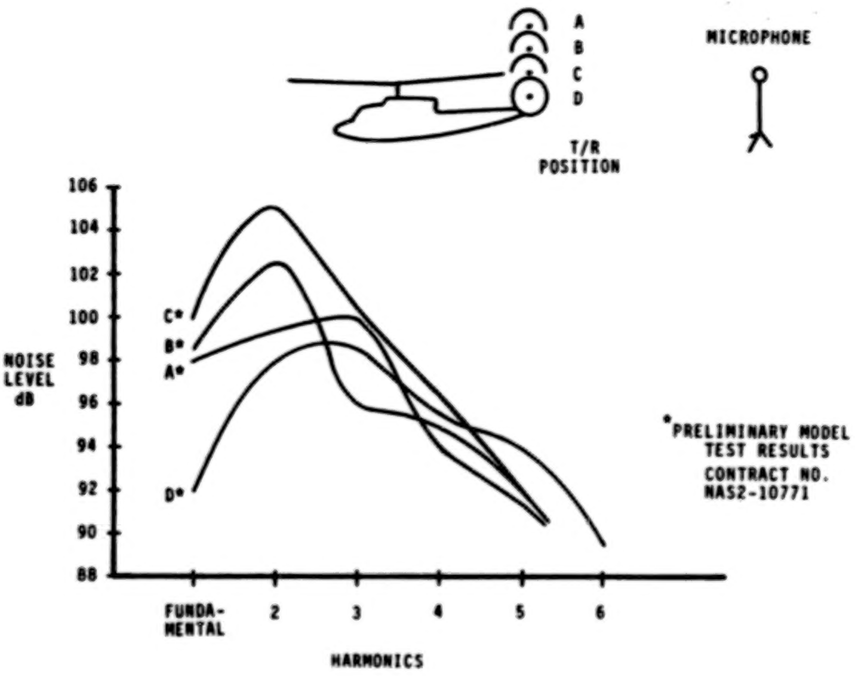
DESIGN EMPHASIS

- CLIMB PERFORMANCE

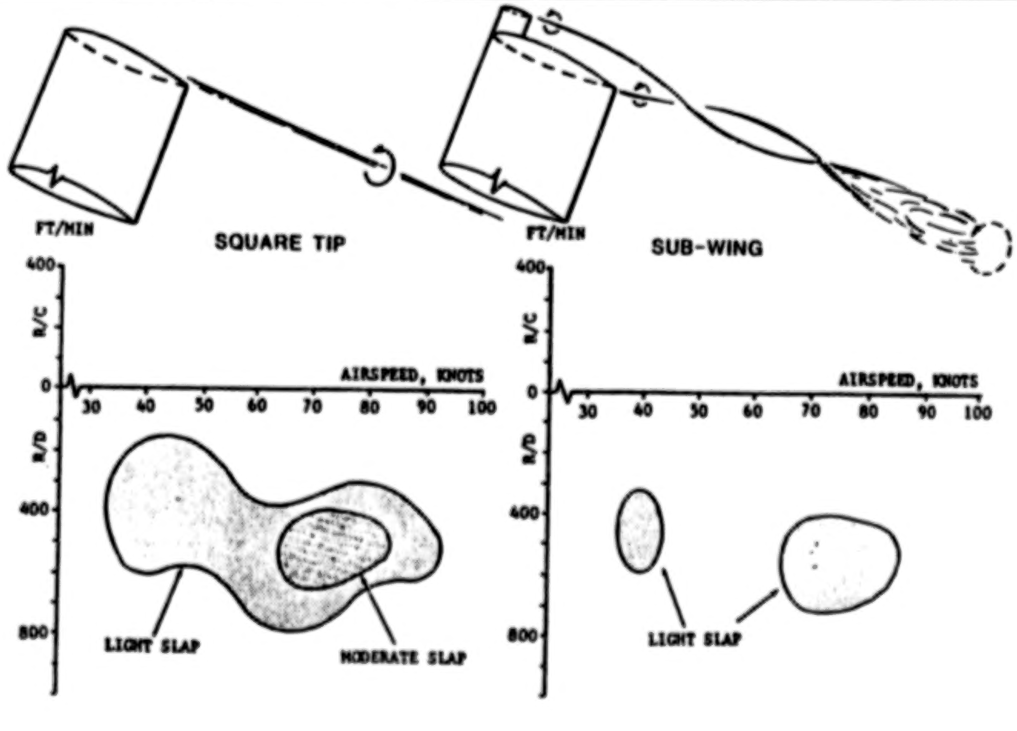
RESEARCH AND TECHNOLOGY

- MAIN ROTOR - TAIL ROTOR INTERFERENCE
- SUBWING
- TRANSONIC BLADE TIP DESIGN

INTERFERENCE

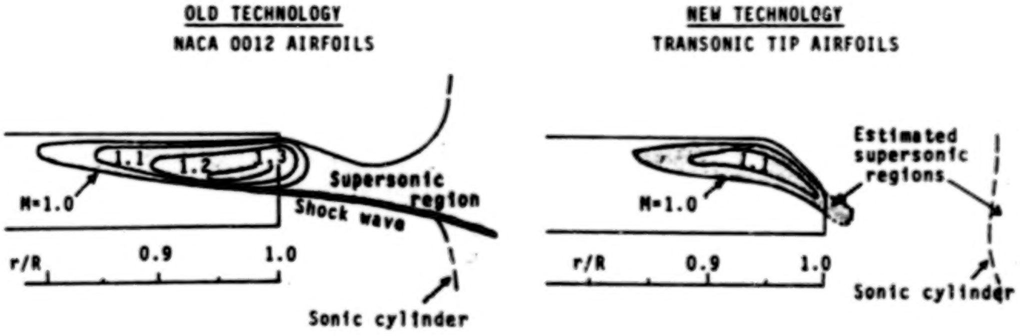


EFFECT OF SUB-WING ON BLADE SLAP



BLADE TIP DESIGN

$M_{1,90}=0.9$ $C_L \approx 0.0$


OLD TECHNOLOGY
NACA 0012 AIRFOILS

NEW TECHNOLOGY
TRANSONIC TIP AIRFOILS

1.1
1.2
1.3
M=1.0
Supersonic region
Shock wave
r/R
0.9
1.0
Sonic cylinder
Estimated supersonic regions
M=1.0
r/R
0.9
1.0
Sonic cylinder

NOISE REDUCTION POTENTIAL = 7 dB

CONCLUSION

- WHAT IS NEEDED IS A PLAN OF ACTION
- THE BASIS FOR THIS PLAN SHOULD BE WORKED OUT DURING THIS WORKSHOP

NEEDED: COOPERATIVE INTERDISCIPLINARY ACTION

OBJECTIVES

- MODIFICATION OF PROPOSED RULES AND REGULATIONS:
  - PROVIDE OPERATIONAL FLEXIBILITY
  - ADDRESS DERIVATIVES
- IMPROVE PREDICTION CAPABILITIES TO MEET RULES
- IMPROVE UNDERSTANDING OF FUNDAMENTALS

COOPERATION

- ESTABLISH CLOSE COORDINATION BETWEEN AGENCIES AND RESEARCH GROUPS
- PROVIDE FOR INDUSTRY INVOLVEMENT IN ALL PHASES

INTERDISCIPLINARY

- BROAD SCOPE:
  - FUNDAMENTALS
  - EMPIRICAL
  - TEST AND CORRELATION
  - OPERATIONS

RESPONSIBILITIES

- ASSIGN RESPONSIBILITIES TO ACHIEVE GOALS ON SCHEDULE

N84 29665

A COMPARISON OF WIND TUNNELS SUITABLE FOR ROTORCRAFT NOISE STUDIES

Paul Soderman
NASA Ames Research Center
Moffett Field, California

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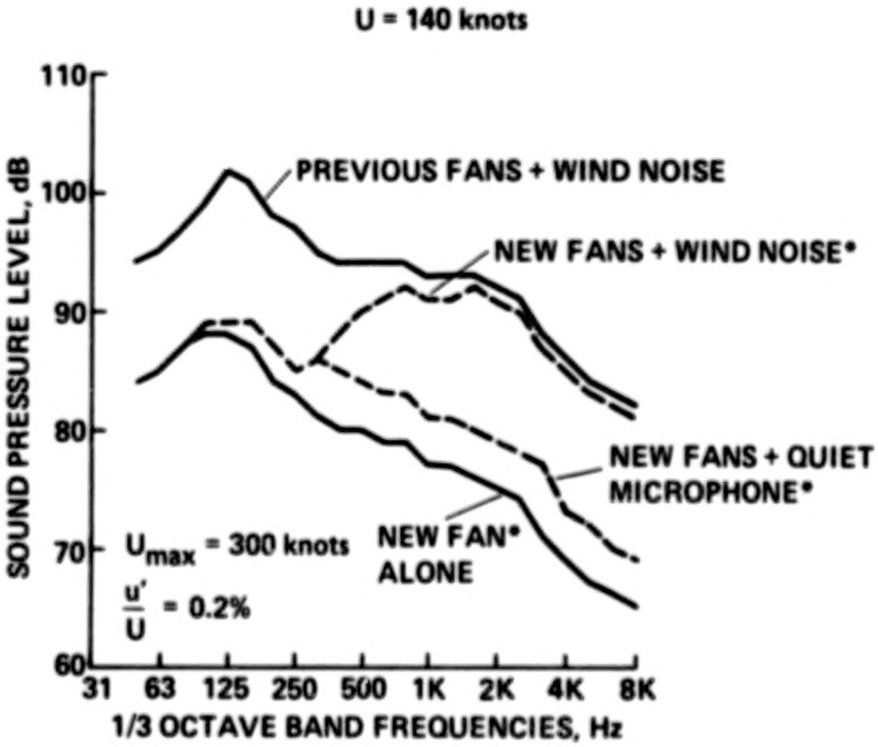
U.S. GOVERNMENT FACILITIES

NASA AMES	TEST SECTION*	PRO	CON
40 x 80 ft	C	• LARGE SCALE • (OR SMALL-SCALE MODEL) • ABSORBENT WALLS	• BACKGROUND NOISE • LOW FREQ. REFLECTIONS
80 x 120 ft	C	• LARGE SCALE	• BACKGROUND NOISE • REFLECTIONS • LOW SPEED (100 knots)
7 x 10 ft #1	O/C	• 3-TEST SECTIONS	• BACKGROUND NOISE • BUFFET AT 120 knots • FLOOR REFLECTIONS • RESTRICTED FAR FIELD
NASA LANGLEY			
4 x 7 m	O	• MEDIUM SCALE	• BACKGROUND NOISE • FLOOR REFLECTIONS
30 x 60 ft	O	• LARGE SCALE	• BACKGROUND NOISE • FLOOR REFLECTIONS • LOW SPEED (90 knots)
ANECHOIC FLOW APPARATUS	O	• ANECHOIC	• SMALL SCALE
NASA LEWIS			
9 x 15 ft	C	• ABSORBENT WALLS	• BACKGROUND NOISE • LOW FREQ. REFLECTIONS • RESTRICTED FAR FIELD
NAVY NSRDC			
8 x 8 ft	O	• QUIET DRIVE SYSTEM	• RESTRICTED FAR FIELD (6 ft FROM JET TO WALL)
ARMY R&T LAB MOFFETT FIELD, CA			
HOVER ROOM	C	• ANECHOIC • REDUCED RECIRCULATION	• HOVER ONLY

*C = CLOSED
O = OPEN

AMES 40- BY 80-FOOT WIND TUNNEL TEST SECTION NOISE

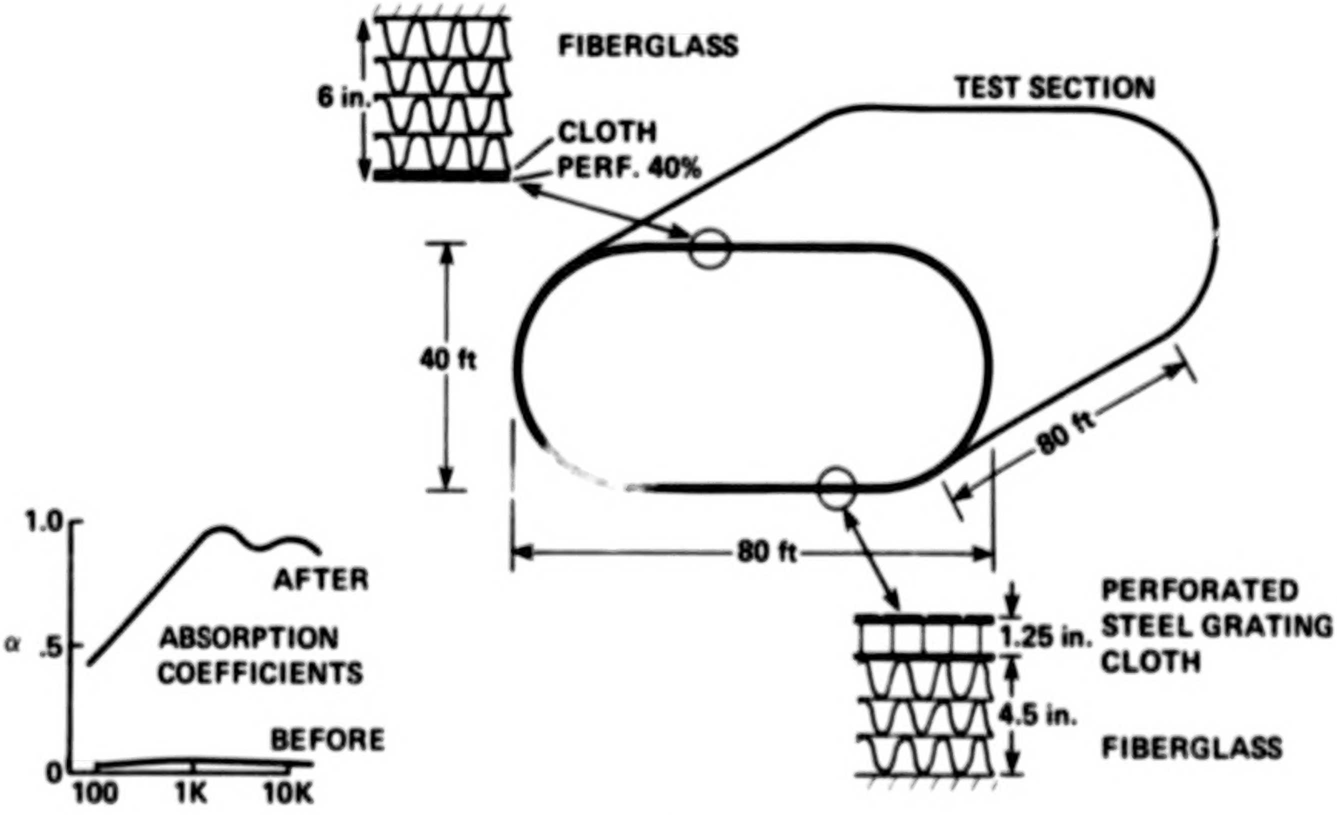
Prior to the Ames 40- by 80-Foot Wind Tunnel modification, the test-section background noise at 140 knots was measured as shown by the upper curve. After the modification, which includes installation of a quieter fan-drive system and one set of acoustically treated corner vanes, the background noise should be reduced considerably. The actual measurements must await the facility start-up in May 1982. However, the estimated background noise is shown on the figure by the lower curve, which represents fan noise only, and the dotted curves, which represent fan and wind noise for a "quiet" microphone and for a microphone on our standard airfoil strut. We feel the strut can be further streamlined so that the final data should fall somewhere between the two dotted curves.



*PREDICTED

NEW 40- BY 80-FOOT WIND TUNNEL ACOUSTIC LINING

The installation of a six-inch-thick acoustic lining on the test section walls as shown is scheduled for June 1982. The quality of acoustic data acquired from aircraft models will be improved immensely. Measured absorption coefficients indicate that the test section should be nearly anechoic at mid- to high-frequencies. Low-frequency reflections will continue to be a problem for rotorcraft rotational noise, but other rotor-noise sources will be easily measured. Small scale models could be used to study simulated low-frequency rotor noise. Conditions will be vastly better than they were during the numerous noise studies previously conducted in this facility.


FIBERGLASS
6 in.
CLOTH
PERF. 40%
TEST SECTION
40 ft
80 ft
80 ft
PERFORATED
STEEL GRATING
1.25 in.
CLOTH
4.5 in.
FIBERGLASS
1.0
α .5
0
100
1K
10K
AFTER
ABSORPTION
COEFFICIENTS
BEFORE

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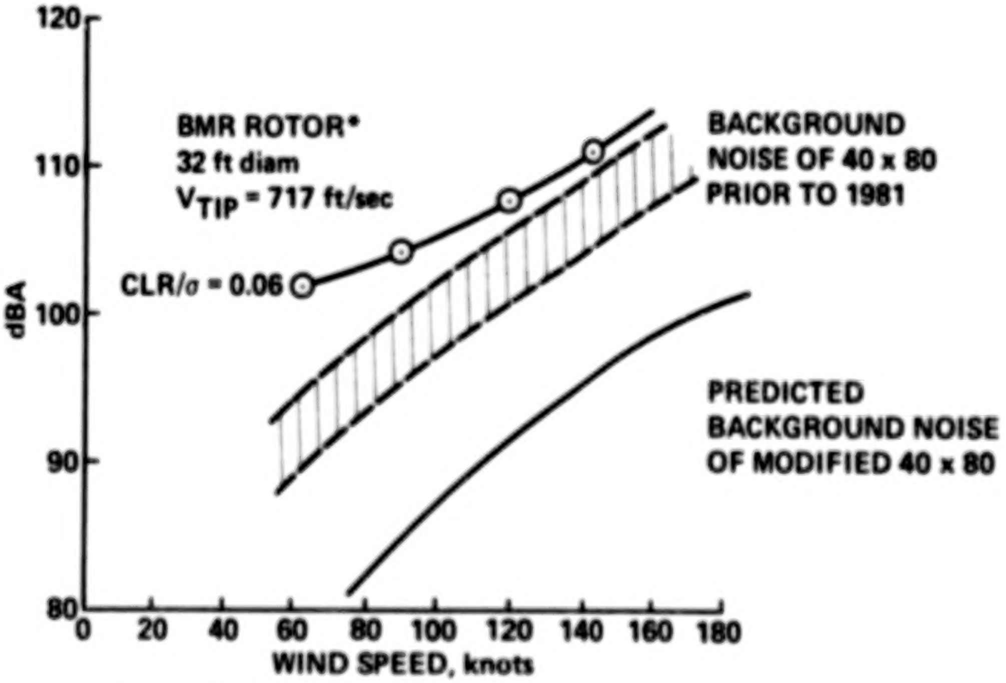
COMPARISON OF ROTOR NOISE AND BACKGROUND NOISE IN THE AMES 40- BY 80-FOOT WIND TUNNEL

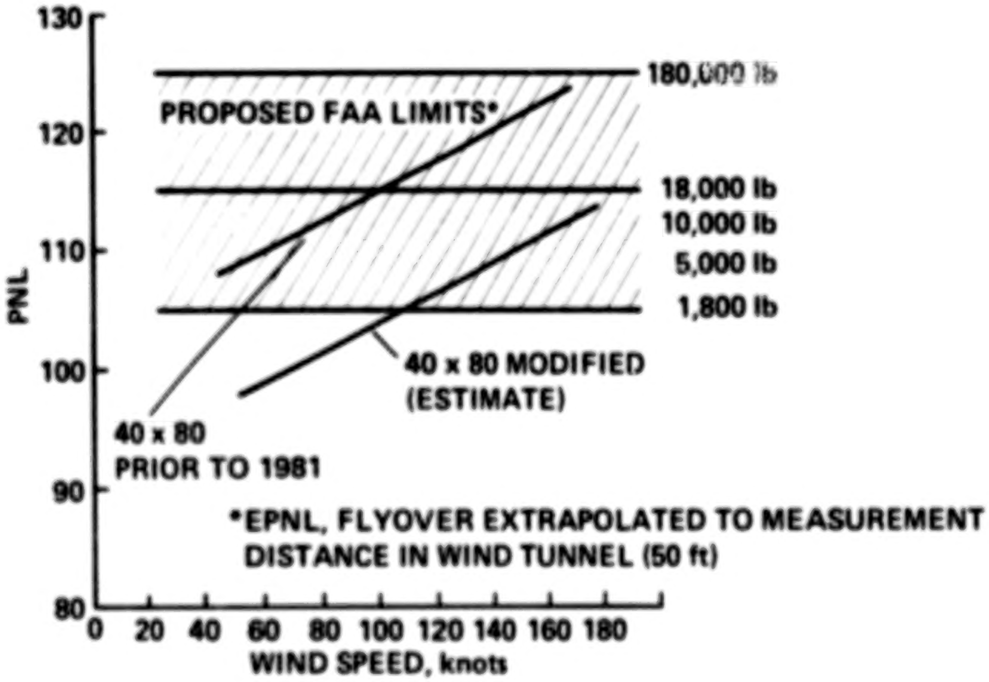
Previous noise data measured from the Bearingless Main Rotor (32 ft diameter, 4 blades) in the 40- by 80-Foot Wind Tunnel were barely above the background noise levels. The figure shows that the separation between that level of rotor noise (in dBA) and the background noise of the modified wind tunnel will be much greater than before.



*AIAA PAPER 81-0092

COMPARISON OF AMES 40- BY 80-FOOT WIND TUNNEL BACKGROUND NOISE AND HELICOPTER NOISE LIMITS

The modified wind-tunnel background noise (in PNL) will be below the proposed FAA noise limits for helicopters except for the lighter helicopters at high speeds. (The proposed limits have recently been withdrawn.) Conceivably, the certification noise levels could be measured in the wind tunnel. Note that the proposed noise limits were corrected to a wind-tunnel measurement distance of 50 ft.



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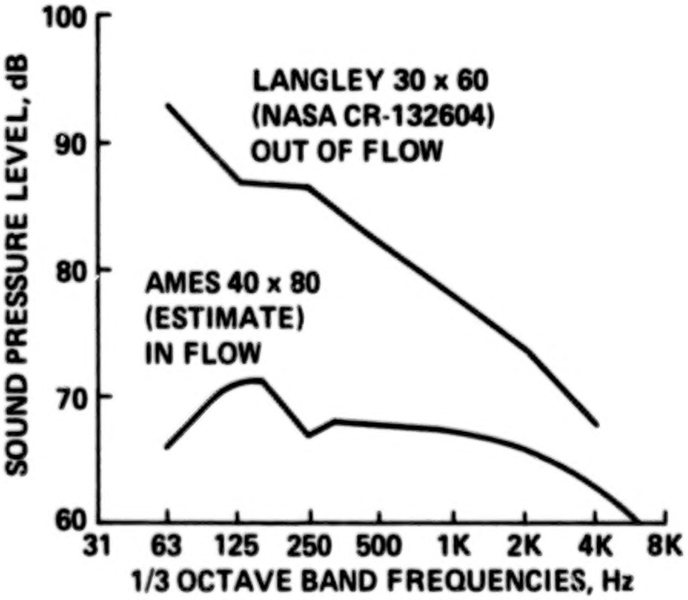
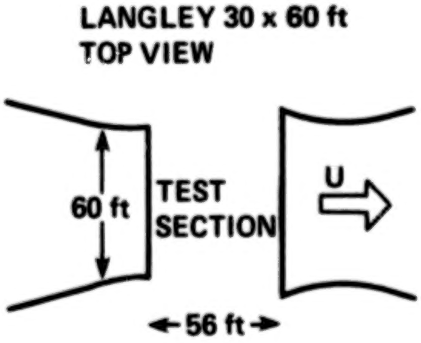
LANGLEY 30- BY 60-FOOT WIND TUNNEL NOISE

The 30- by 60-Foot Wind Tunnel background noise is considerably higher than the estimated 40- by 80-Foot Wind Tunnel noise. No doubt this is due to the fact that the drive fan is very close to the test section. Nonetheless, this facility has been used successfully for certain aircraft noise studies. The large open test section is obviously an advantage with regard to microphone placement. The top speed of this tunnel is approximately 90 kts.

BACKGROUND NOISE

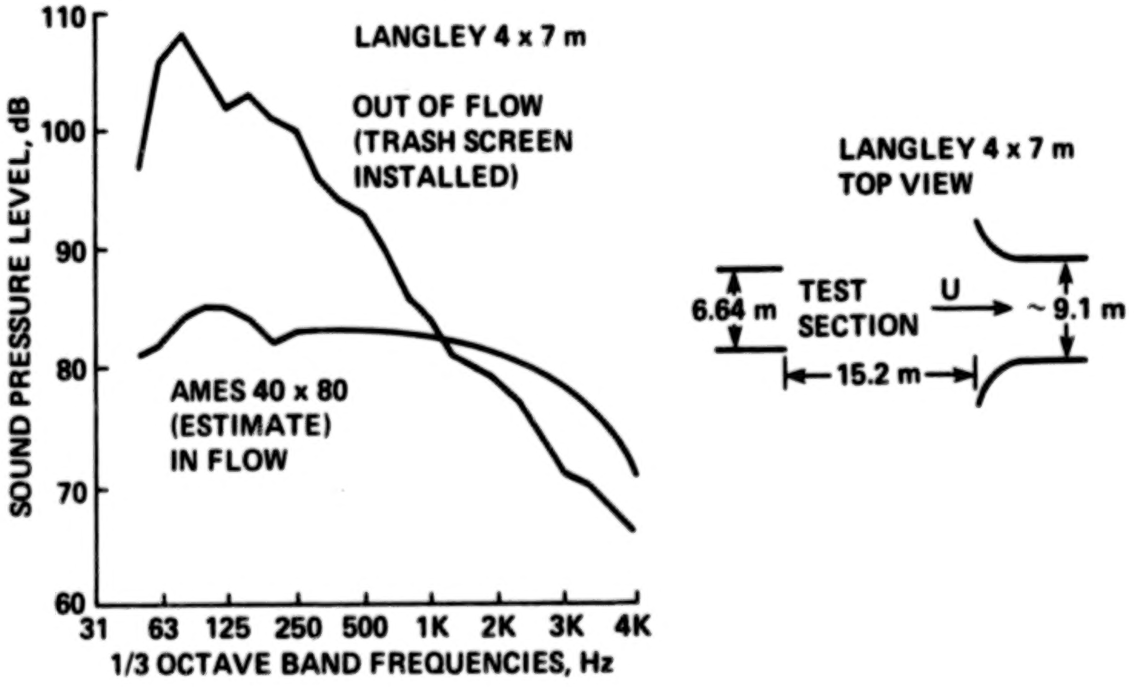
U = 65 knots





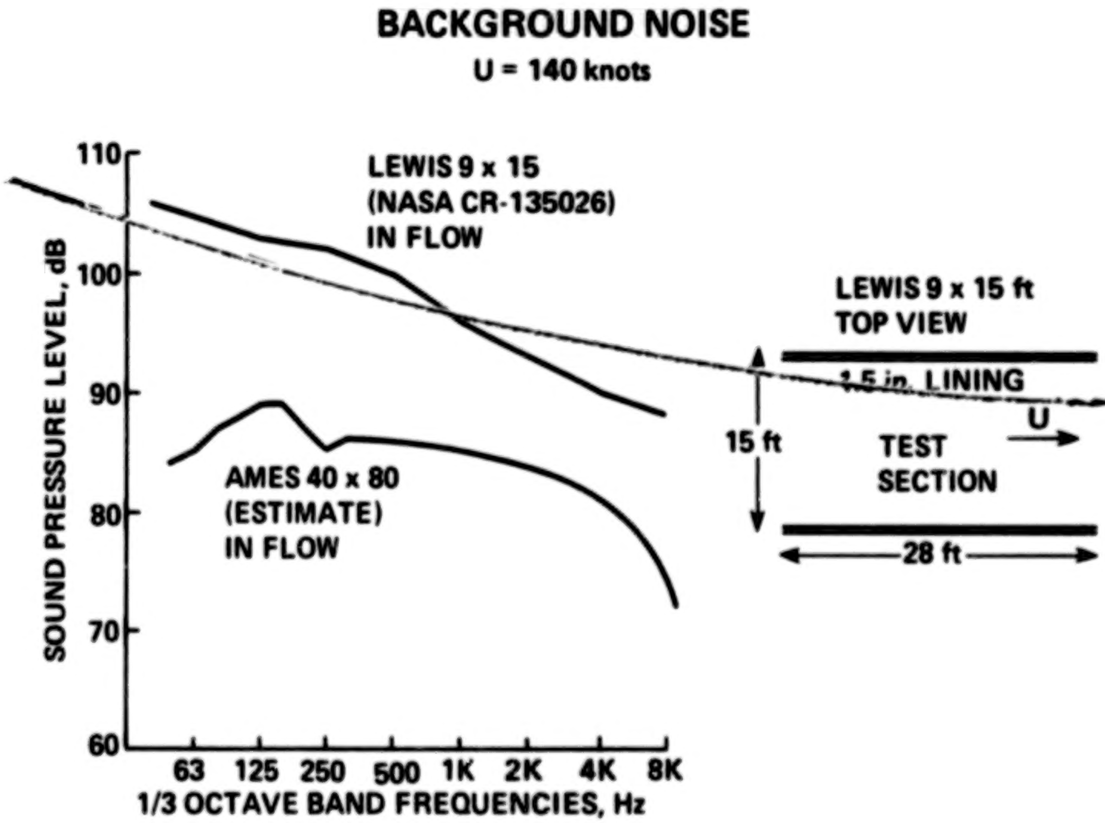
LANGLEY 4- BY 7-METER WIND TUNNEL NOISE

Recent acoustic measurements near the 4- by 7-meter wind tunnel clean test section show that the mid- to high-frequency noise is reasonably low, but there is considerable low-frequency noise due to the drive fan. However, the fan noise should be substantially reduced after the planned trash screen removal, since the screen is a high loss component of the wind tunnel. This unique wind tunnel, with its large, open test section, is a valuable facility for studying rotorcraft models with sufficiently high noise signatures.


BACKGROUND NOISE
U = 125 knots

LEWIS 9- BY 15-FOOT WIND TUNNEL NOISE

This acoustically treated wind tunnel has been used primarily for propeller noise work. It is conceivable that it could also be used for small-scale rotorcraft noise studies. However, the application would be restricted by: a) the background noise, b) the thin acoustic lining (1.5 inch), which is ineffective below 500 Hz, and c) the closed test section, which constrains microphone placement.

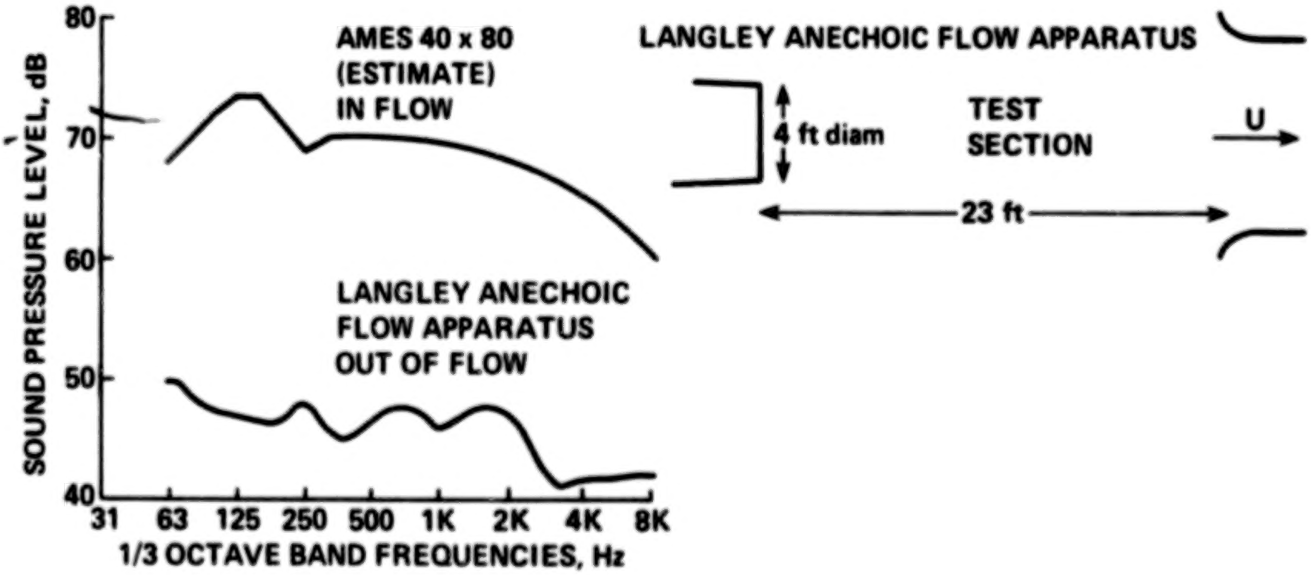


LANGLEY ANECHOIC FLOW APPARATUS

The Langley Anechoic Flow Apparatus is a free jet which issues vertically from the floor of a 7- by 10- by 10-m anechoic chamber in the Aircraft Noise Reduction Laboratory. The background noise is extremely low as shown in the figure. With the 4-ft-diameter nozzle, the maximum flow speed is 71 kts. This facility is well suited for small-scale rotorcraft noise studies.

BACKGROUND NOISE

U = 71 knots



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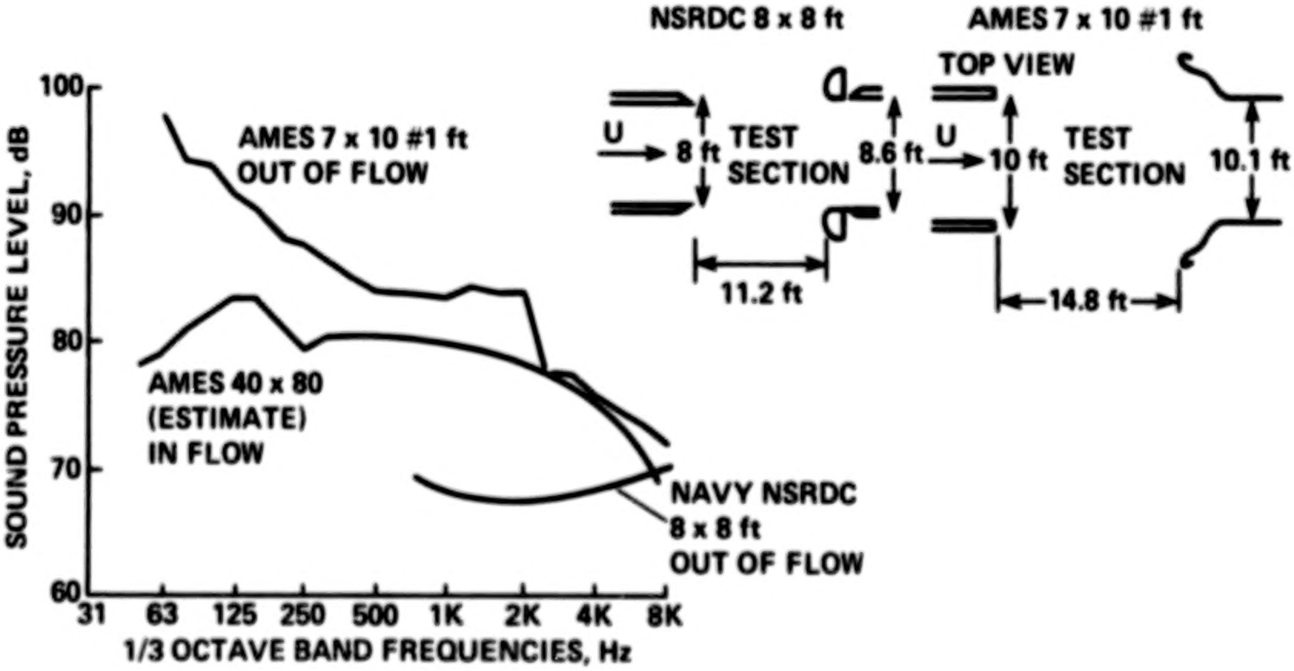
COMPARISON OF AMES 7- BY 10-FOOT #1 AND NAVY 8- BY 8-FOOT WIND TUNNEL NOISE

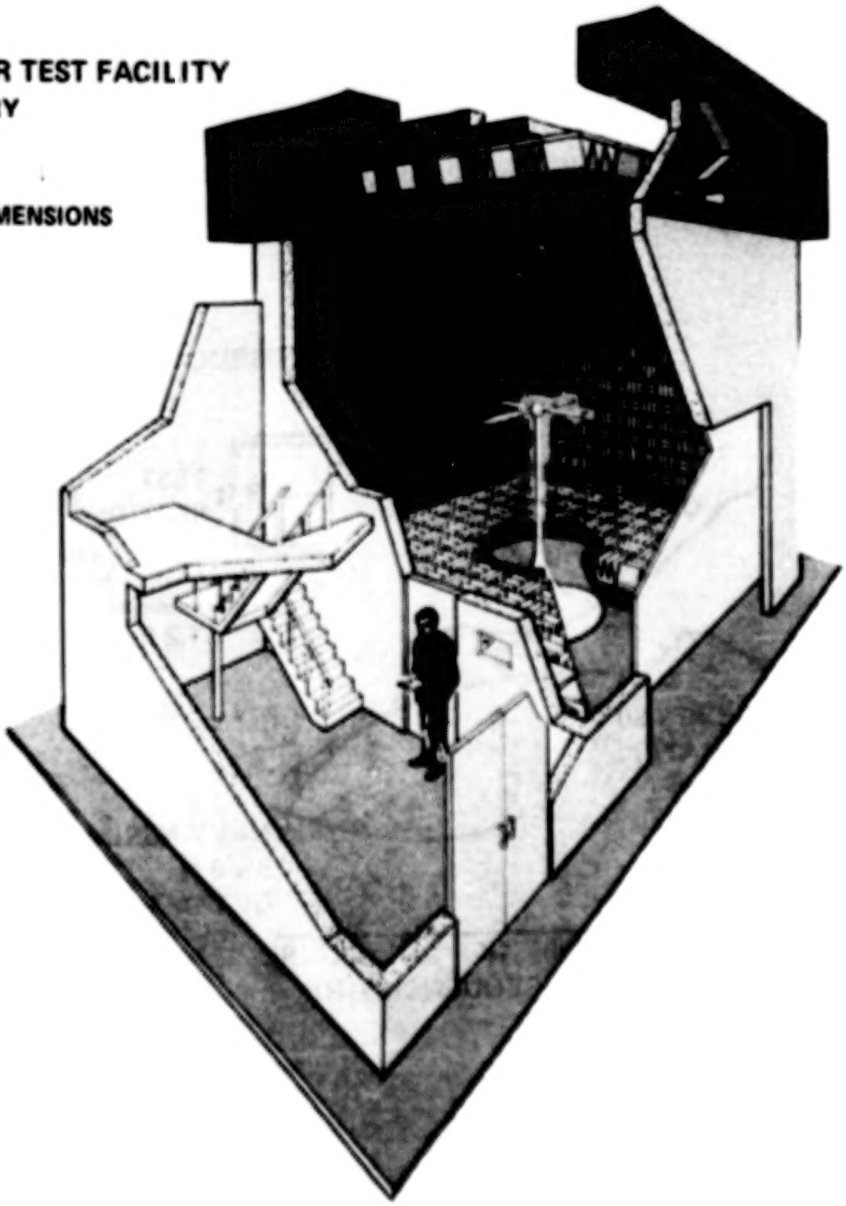
The Navy 8- by 8-Foot Wind Tunnel is extremely quiet due to well-designed duct silencers. The 8-ft-square open test section would be suitable for small-scale rotorcraft testing. Unfortunately, the rather small measurement room (17 × 19-1/2 × 19-1/2 ft working dimensions) and short test section restrict the measurement field. The Ames 7 × 10 #1 is noisier than the Navy facility, but quieter than many others due to a fan silencer. The measurement room is large, but the collector position restricts certain microphone positions. The 7 × 10 can also be operated with a closed, hard or a closed, absorbent test section.

BACKGROUND NOISE
U = 109 knots



ARMY ANECHOIC HOVER CHAMBER

Although not a wind tunnel, this nicely designed anechoic chamber has intake and exhaust ports which allow air to flow through a hovering rotor with minimum recirculation. Foam wedges, 26 inches long and held 4 inches from the wall, make the room anechoic down to 110 Hz. The floor can be raised for easy access to the drive stand and rotor. The room makes a good laboratory for detailed optical and acoustical probing of rotor blades in hover.



ANECHOIC ROTOR HOVER TEST FACILITY
U.S. ARMY R & T LABORATORY
MOFFETT FIELD, CA

26 BY 32 BY 28 ft OUTSIDE DIMENSIONS

ANECHOIC TO 110 Hz

NON-U.S.-GOVERNMENT FACILITIES

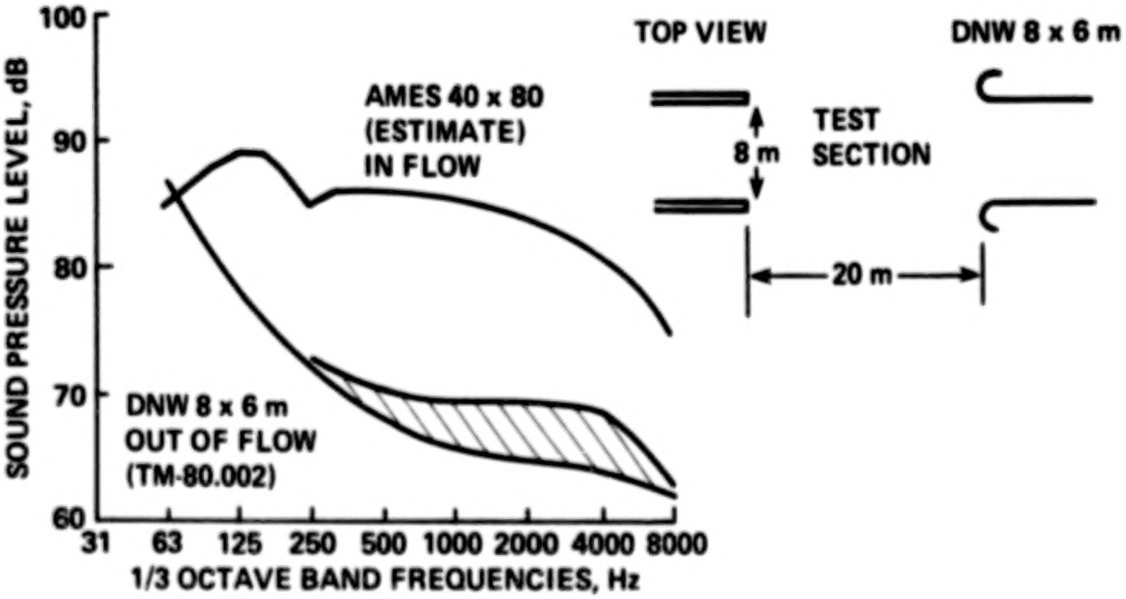
		TEST SECTION*	PRO	CON
DNW	8 x 6 m	O	• QUIET • LARGE SCALE • ANECHOIC	• COST (?)
RAE	24 ft	O	• LARGE SCALE	• NOISY
ONERA S1	8 m diam	C	• LARGE SCALE • TRANSONIC	• NOISY
CEPRA 19	3 m diam	O	• QUIET	• SMALL SCALE
BBN	4 x 4 ft	O	• QUIET	• SMALL SCALE
BOEING VERTOL	20 x 20 ft	C/O	• LARGE SCALE	• NOISY • REFLECTIONS
MIT	5 x 7.5 ft	O	• QUIET	• SMALL SCALE
UTRC	21" x 31" 50" diam	O	• QUIET	• SMALL SCALE

*C = CLOSED
O = OPEN

GERMAN-DUTCH WIND TUNNEL (DNW) NOISE

The limited measurements of background noise taken to date in DNW indicate that this is a very quiet facility. Unlike most other wind tunnels, DNW was designed from the beginning to be an aeroacoustic facility. With its large, open test section (8 × 6 m) and acoustically treated test chamber, it is ideal for rotorcraft noise studies. The curve shown here is from "Aeroacoustic Checkout of DNW", DNW-TM-80.002, February 1980. It is anticipated that more detailed data will be released in the near future.

BACKGROUND NOISE
U = 140 knots



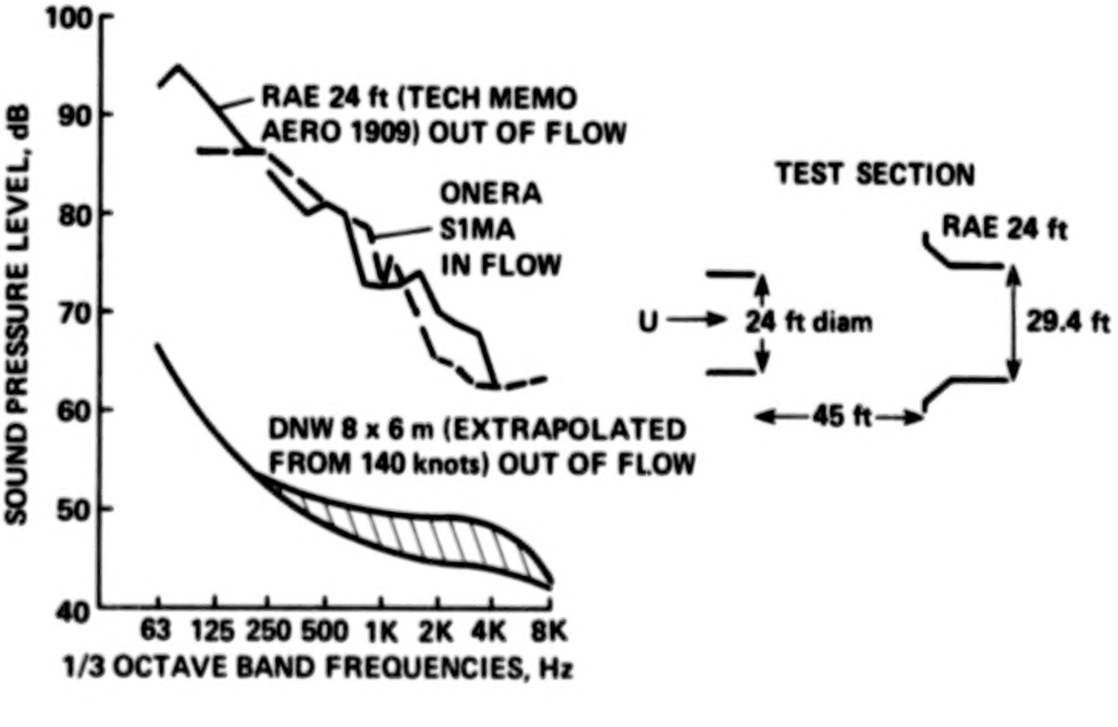
RAE 24-FOOT AND ONERA S1MA WIND TUNNEL NOISE

The background noise of the RAE 24-Foot Wind Tunnel is high because the drive fan is in line-of-sight of the test section. The measurement of low-frequency rotor noise would be difficult. However, the facility has been used at low speeds for propeller noise work. Furthermore, a study of tail-rotor noise in this facility is planned as part of a cooperative program between the RAE and NASA Ames.

The ONERA S1MA wind tunnel has a closed, 8-meter-diameter test section that can be lined with a 5-inch-thick absorbent lining. This facility is transonic and has been used successfully for a jet-noise study conducted by ONERA and NASA Ames. It could also be used for certain rotorcraft noise work since the background noise is similar to the RAE 24-foot despite the fact that the microphone must be in the flow.

BACKGROUND NOISE

U = 65 knots



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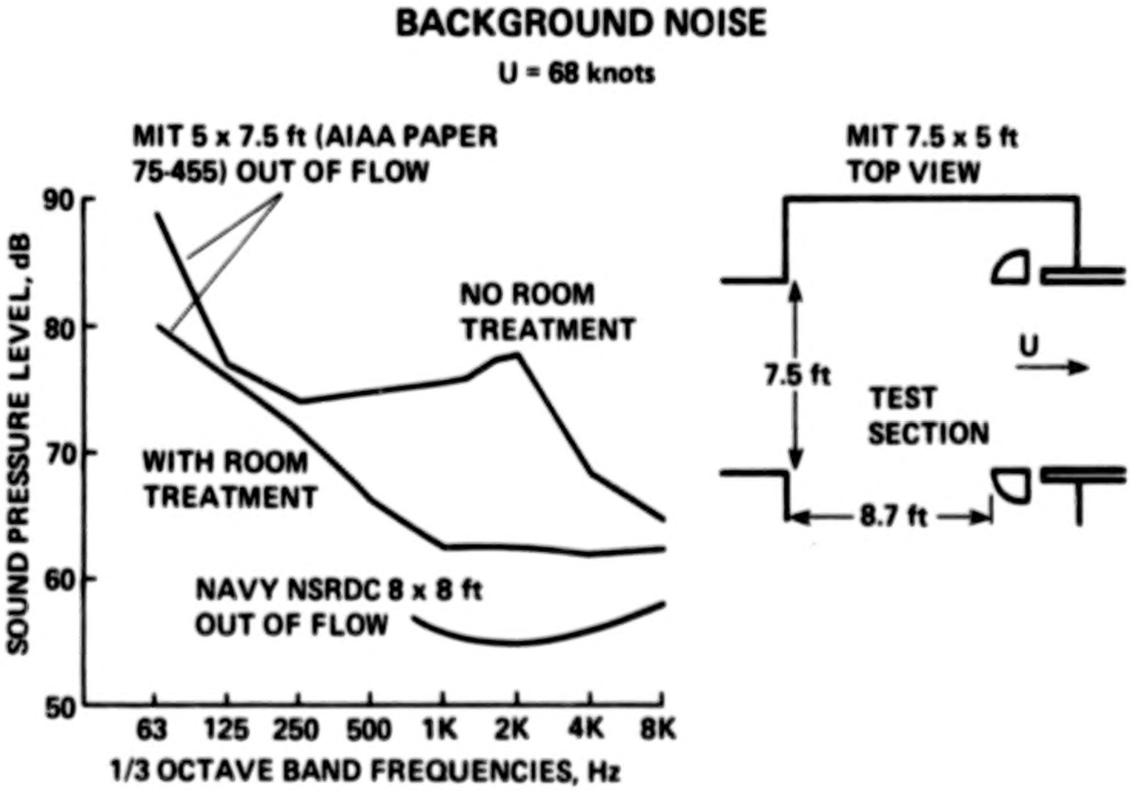
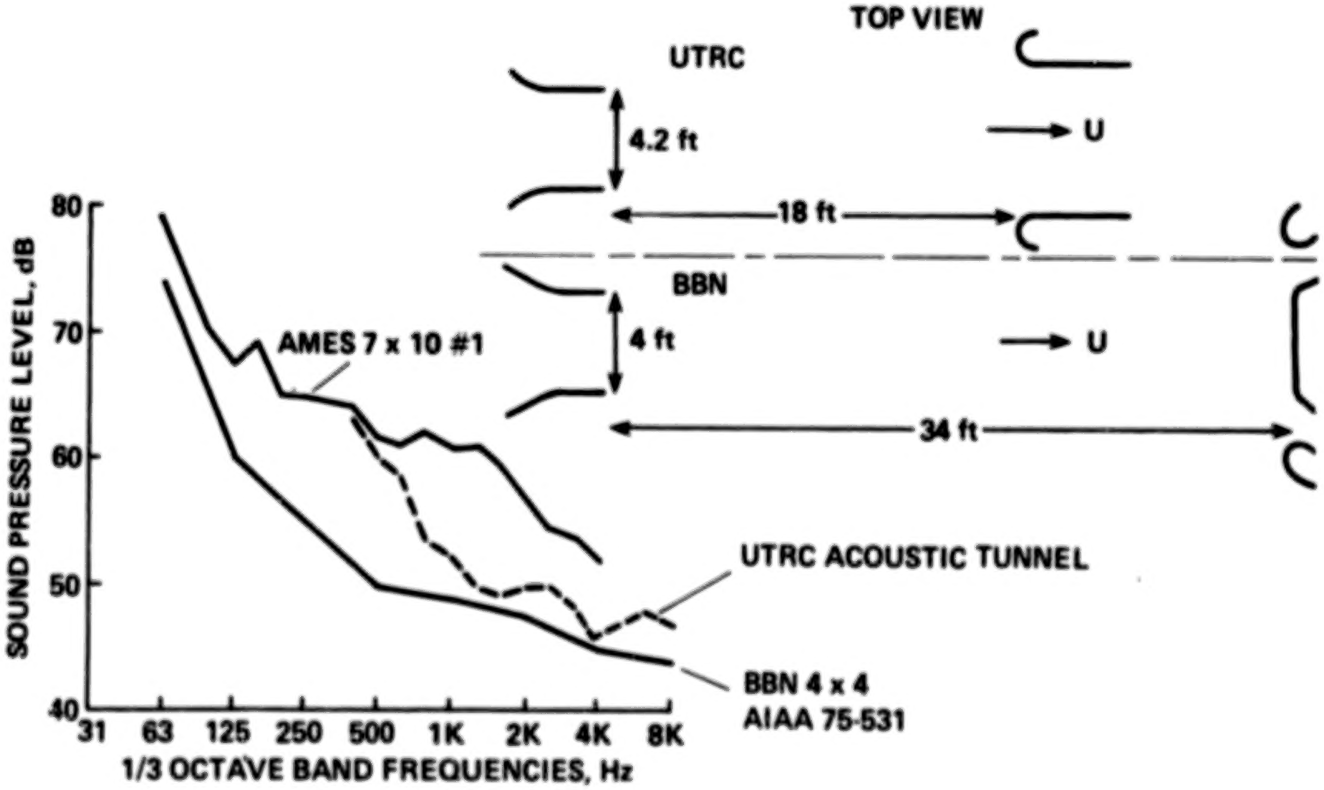
MIT 7.5- BY 5-FOOT WIND TUNNEL NOISE

The Massachussetts Institute of Technology has a small, open-jet wind tunnel for rotorcraft noise research. The facility is very quiet as shown here. The data illustrate that acoustically treating the test chamber serves not only to create an anechoic environment, but also to reduce the background noise. It would appear that the collector and test chamber wall restrict the microphone placement. Nonetheless, the facility has been used successfully for numerous small-scale rotor-noise studies.



A COMPARISON OF THREE SMALL WIND TUNNELS

Both the UTRC Acoustic Tunnel and the BBN 4- by 4-Foot Wind Tunnel are low-noise facilities, somewhat quieter than the Ames 7- by 10-Foot Wind Tunnel No. 1. The UTRC and BBN test sections are longer than most, and are surrounded by anechoic chambers. Nozzle size can be traded off for airspeed. For example, the UTRC 50-inch-diameter nozzle with a maximum Mach number of 0.15 can be replaced by a 21- by 31-inch nozzle with a maximum Mach number of 0.55. The small section has been used to good purpose for isolated airfoil studies.



BACKGROUND NOISE

U = 68 knots

CONCLUSIONS

- MANY GOOD SMALL-SCALE FACILITIES ARE AVAILABLE FOR ROTORCRAFT NOISE RESEARCH
- AMES 40 x 80/80 x 120 PROBABLY WILL BE GOOD LARGE-SCALE FACILITY FOR ACOUSTIC RESEARCH OF LARGE OR SMALL ROTORCRAFT, BUT ACOUSTIC QUALITY OF MODIFIED FACILITY HAS NOT YET BEEN MEASURED
- DNW IS PROBABLY BEST AEROACOUSTIC FACILITY, BUT MAY BE EXPENSIVE
- OTHER PARAMETERS TO CONSIDER BESIDES BACKGROUND NOISE:
  - TURBULENCE LEVEL AND SCALE
  - RESTRICTED MEASUREMENT FIELD
  - LOCAL REFLECTIONS
  - SPEED RANGE
- ADVANCED MEASUREMENT TECHNIQUES CAN IMPROVE DATA QUALITY FROM ANY FACILITY

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NOISE REDUCTION EXPERIENCE
AT HUGHES HELICOPTERS, INC.

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QUIET HELICOPTER PROGRAM

Relevant noise reduction experience at Hughes Helicopters, Inc. dates back to 1969, when, under the sponsorship of U.S. Army and DARPA, the Quiet Helicopter Program was initiated. The objective of this program was to reduce the noise signature of a light observation helicopter OH-6A (fig. 1) irrespective of performance penalties and measure the quietest operation. The program was conducted in two phases.

Figure 1

QUIET HELICOPTER PROGRAM - PHASE I

It was realized that the tail rotor was the primary noise source on OH-6A and quieting it was one of the main objectives of the Phase I program. The noise reduction in this phase was conducted in two steps. First, tail rotor tip speed was reduced from 672 fps to 514 fps with an accompanying increase in solidity. Two tail rotors, one a two-bladed wide chord tail rotor (twice the chord size of the OH-6A) and the other a four-bladed tail rotor (phasing 60°/120°), were tested. Main rotor rpm was reduced from 100% N_2 to 70% N_2. Flyover (85 kts) and hover tests were conducted. Tests showed that with no reduction in performance (2400 lbs G.W. 100% N_2) compared to the standard two-bladed tail rotor of the OH-6A, use of the four-bladed x and two-bladed wide chord tail rotors at reduced tip speeds resulted in OASPL reductions of 4.5 dB and 2.0 dB respectively. With reduction in performance (reduction of gross weight from 2400 lbs to 1450 lbs), main rotor tip speed reduction (70% N_2) resulted in OASPL reductions of 10.5, 13.0 and 11.5 dB with standard, four-bladed x and two bladed wide-chord tail rotors respectively. Main and tail rotor noise reductions in this program changed the OH-6A principal noise signature sources from rotor to non-rotor as shown in figure 2.

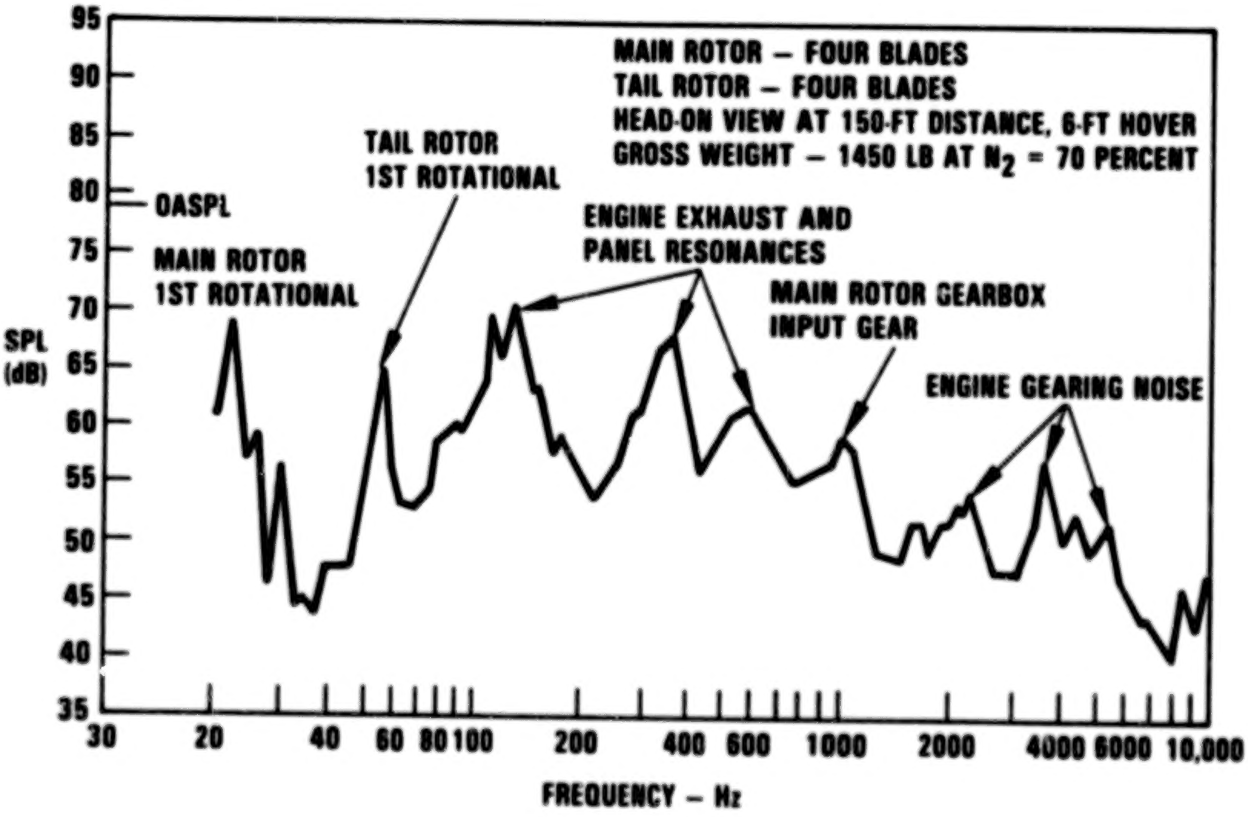

Figure 2

QUIET HELICOPTER PROGRAM - PHASE II (THE QUIET ONE)

The objective of this phase was to further reduce the noise signature of the OH-6A over that achieved in Phase I by extensive modifications to the aircraft and operation at 67% N_2 and 1600 lbs gross weight. The modifications made were: a five-bladed main rotor with trapezoidal tips, main rotor tip speed reduction from 648 fps (100% N_2) to 433 fps (67% N_2), a four-bladed staggered tail rotor with blade phasing of 75°/105°, tail rotor tip speed reduction from 672 fps to 360 fps, a noise suppressor for the engine exhaust, modified gearing for the main-rotor gearbox, damping material added to some of the shafting and gearing, and acoustic blanket material applied to transmission and engine compartment. The aircraft with all these features installed was called "The Quiet One" (fig. 3).

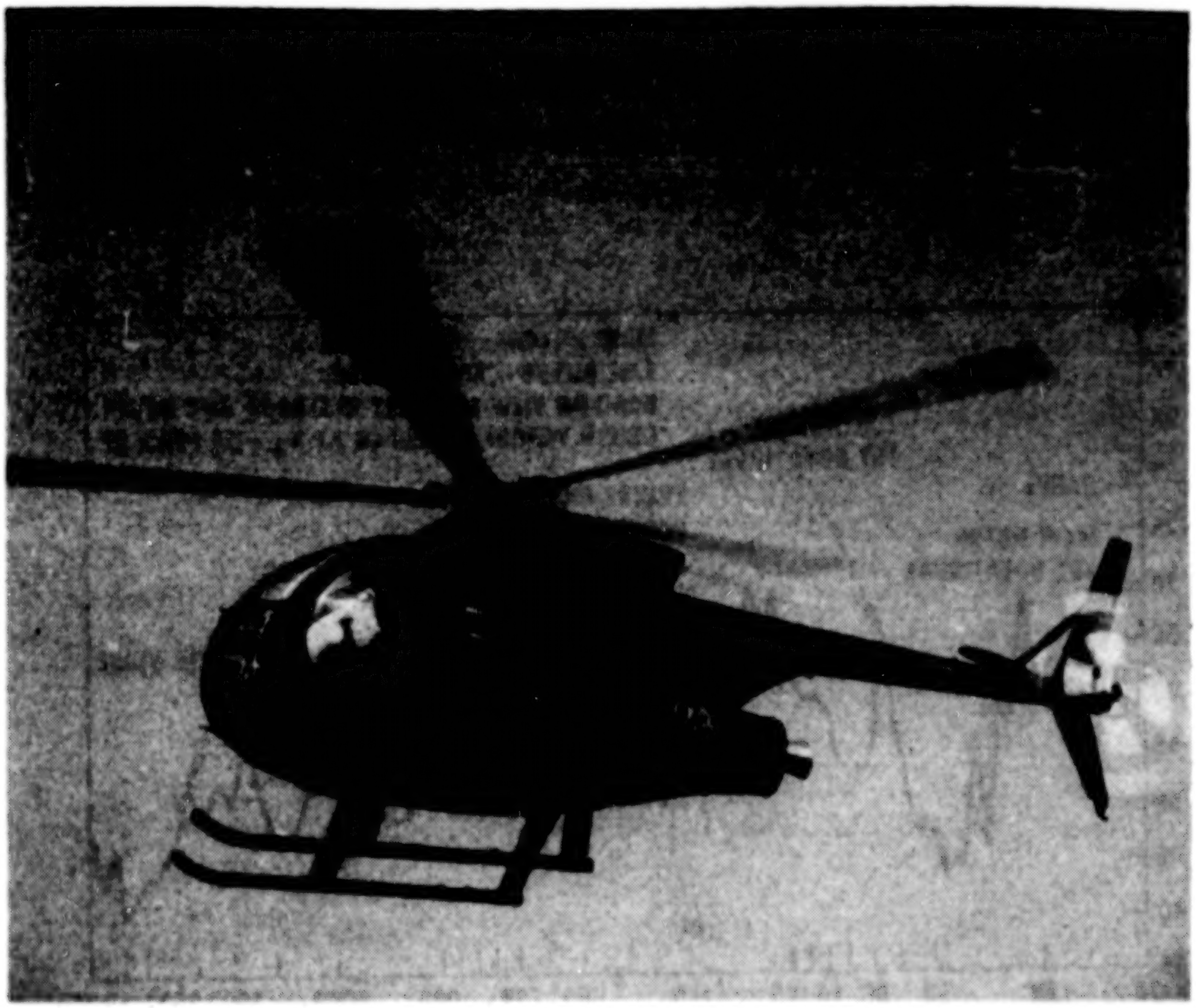

Figure 3

QUIET HELICOPTER PROGRAM - PHASE II - RESULTS (HOVER)

Hover flight tests were conducted at 6-ft skid height with the noise data recorded at 0-deg azimuth heading 150 ft away. Figure 4 shows the hover noise spectra of a standard OH-6A at 1450 lbs gross weight and 100% N_2 and "The Quiet One" at 1600 lbs gross weight and 67% N_2. Substantial noise reduction could be seen throughout the frequency range. The figure shows a 20 dB decrease in the first-harmonic rotational noise of the main rotor and 30 dB decrease in the first-harmonic rotational noise of the tail rotor with the Phase II quiet helicopter (the "Quiet One"). The exhaust pipe resonance at 125 Hz and exhaust noise between 300 and 1000 Hz have also been reduced. It was found that a 17 dB reduction in OASPL was achieved at 1600 lbs gross weight while the reduction was about 14 dB OASPL at 2400 lbs gross weight. More details of the Phase II helicopter and the flight tests are given in reference 1.

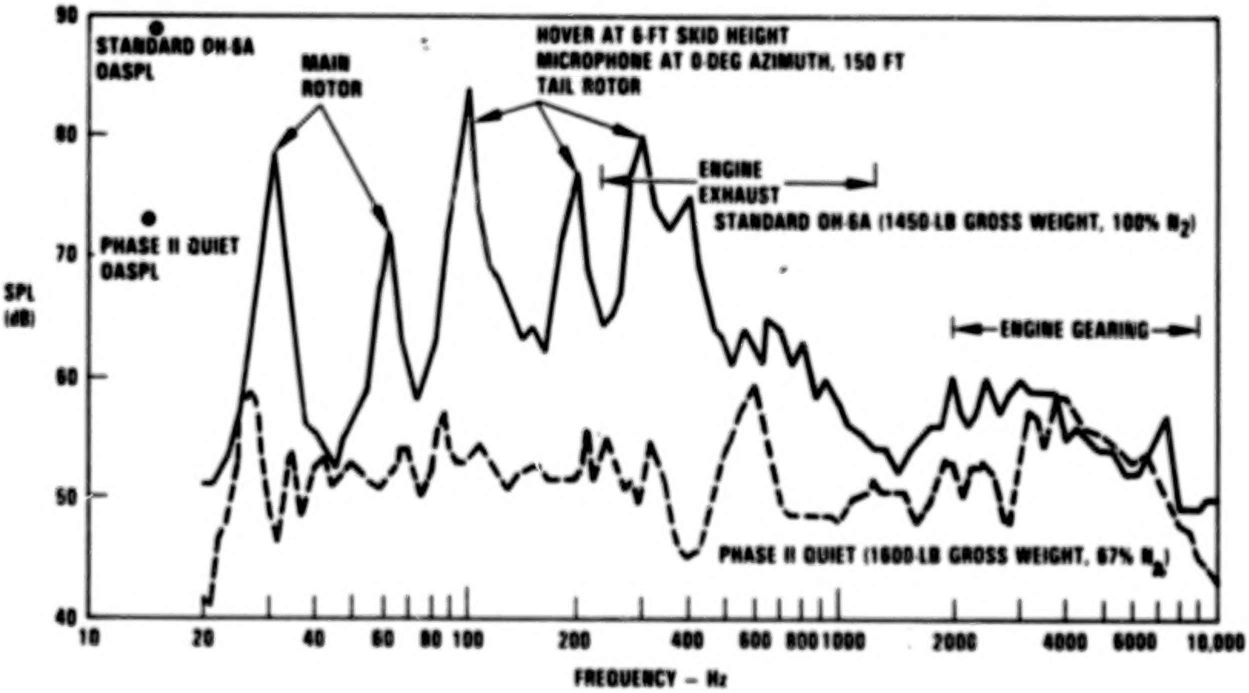

Figure 4

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QUIET HELICOPTER PROGRAM - PHASE II - RESULTS (FLYOVER)

For flyover tests, the maximum speed was limited to 70 kts. The aircraft were flown at an altitude of 100 ft. Figure 5 shows the 1/10 octave band spectra of a standard OH-6A and the Phase II quiet helicopter for the overhead microphone position (nearest to the helicopters during flyover). Appreciable reductions in noise are seen across the entire spectrum with a 15 dB reduction in OASPL. Concurrently with instrumented noise measurements, a series of stop watch measurements (ref. 1) were made at low ambient noise conditions to determine the aural detection distances of the standard OH-6A and Phase II quiet helicopter during 70 kt flyover at 100 ft altitude. It was found that the average aural detection range of the standard OH-6A was 6 to 7 times greater than that of the Quiet Helicopter, even when operating both aircraft at 2400 lbs, 100% N_2 and 120 kts.

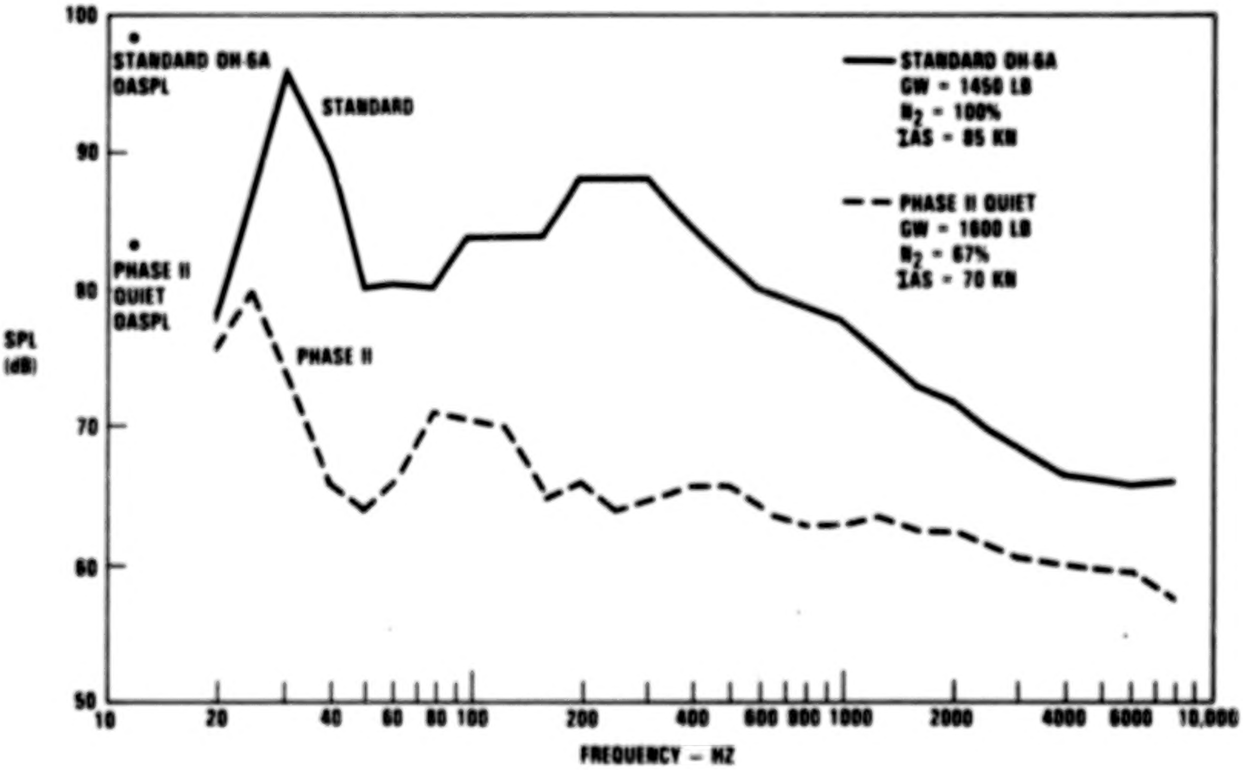

Figure 5

QUIET HELICOPTER PROGRAM - FLYOVER TESTS (TIME HISTORIES)

Sample time histories of the overall sound pressure levels of the standard OH-6A and quiet helicopters during a flyover at 40 kts at an altitude of 100 ft are shown in figure 6. These flyover tests were conducted by NASA Langley and are described in detail in reference 2. For the two helicopters, the sound pressure levels increase as the aircraft approaches the microphone position, reaches the maximum near the overhead position and decreases as the helicopter passes beyond the microphone position. The two shapes are similar except that the maximum sound pressure level of the Quiet Helicopter is 10 dB lower and the warning time is much less. Both aircraft were flown at a gross weight of 1600 lbs.

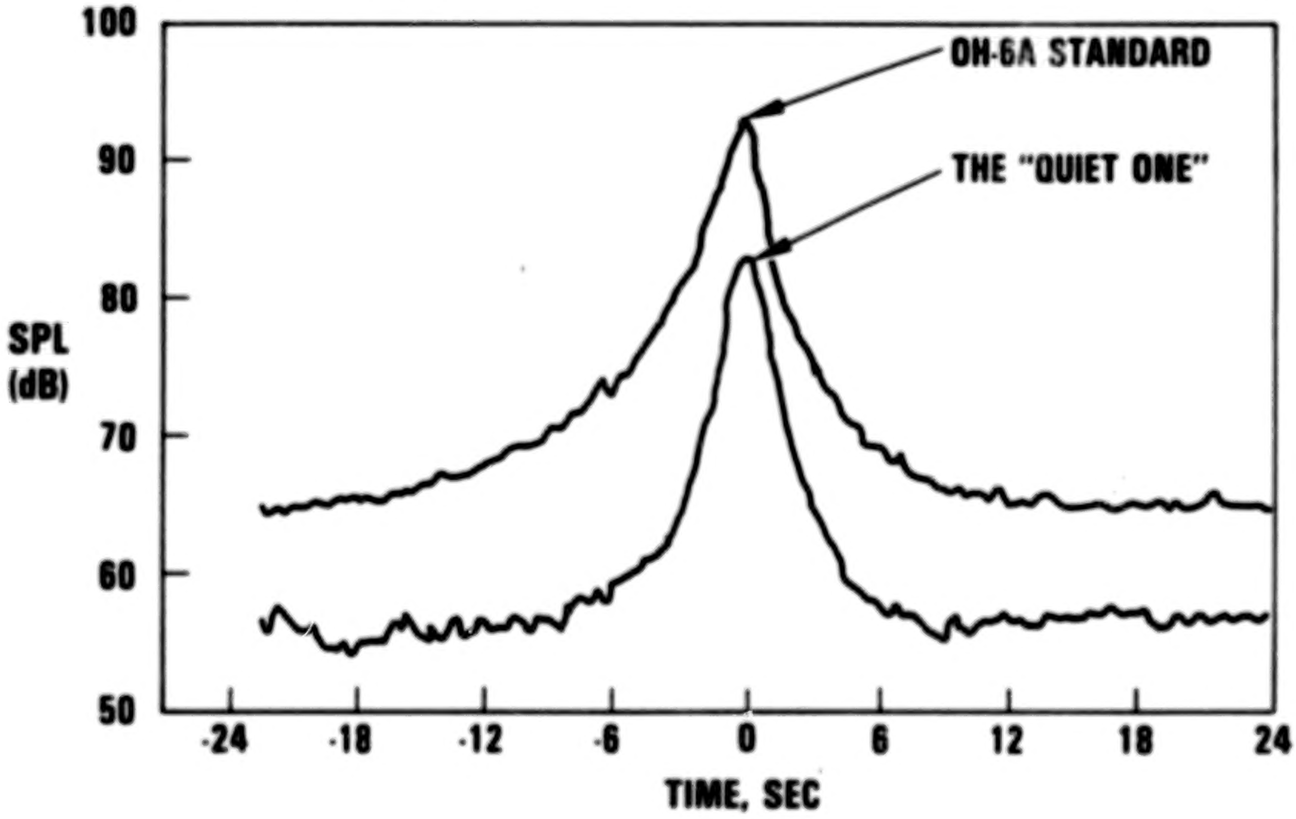

Figure 6

QUIET HELICOPTER PROGRAM - PHASE II (PENALTIES)

The extensive modifications of the Phase II Quiet Helicopter resulted in performance degradation. The operational gross weight of the Quiet Helicopter was limited to 1600 lbs (compared to 2400 lbs for the standard OH-6A) and its maximum forward speed was limited to 70 kts (compared to 100 kts for the standard OH-6A). The empty weight of the helicopter was increased by 192 lbs (8% of gross) with all the modifications. Autorotation performance of the Quiet Helicopter was poor and therefore it was not an operationally capable helicopter. Nevertheless the Quiet Helicopter demonstrated the technical feasibility of achieving large noise reductions using well known noise reduction techniques such as rotor tip speed reduction, engine exhaust muffling, acoustic blanketing, etc.

- Penalties
    - Gross weight reduction 2400 to 1600 lbs
    - Forward speed reduction 100 to 70 kts
    - Empty weight increase + 192 lbs
    - Autorotation performance poor
- Not an operationally feasible helicopter

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QUIET HELICOPTER - HOVER TESTS - AMPLITUDE MODULATIONS

Figure 7 shows the 1/3 octave band spectrum of the Quiet Helicopter in hover at 6 ft skid height. The data was obtained for 0 deg azimuth heading 200 ft away. The sound pressure levels were D-weighted to simulate the subjective annoyance of human beings. As shown in figure 7, for the Quiet Helicopter, at least on a D-weighted scale, the importance of non-rotor noise sources such as engine exhaust, engine gearing etc. is obvious. It is also seen in figure 7 that the sound pressure levels in all the 1/3 octave bands modulated at 1/rev, 5/rev and 10/rev frequencies. It is believed that the rotor noise heard is actually other noise sources possibly being modulated at the fundamental frequencies or harmonics of the main rotor. The cause of these strong amplitude modulations is not identified.

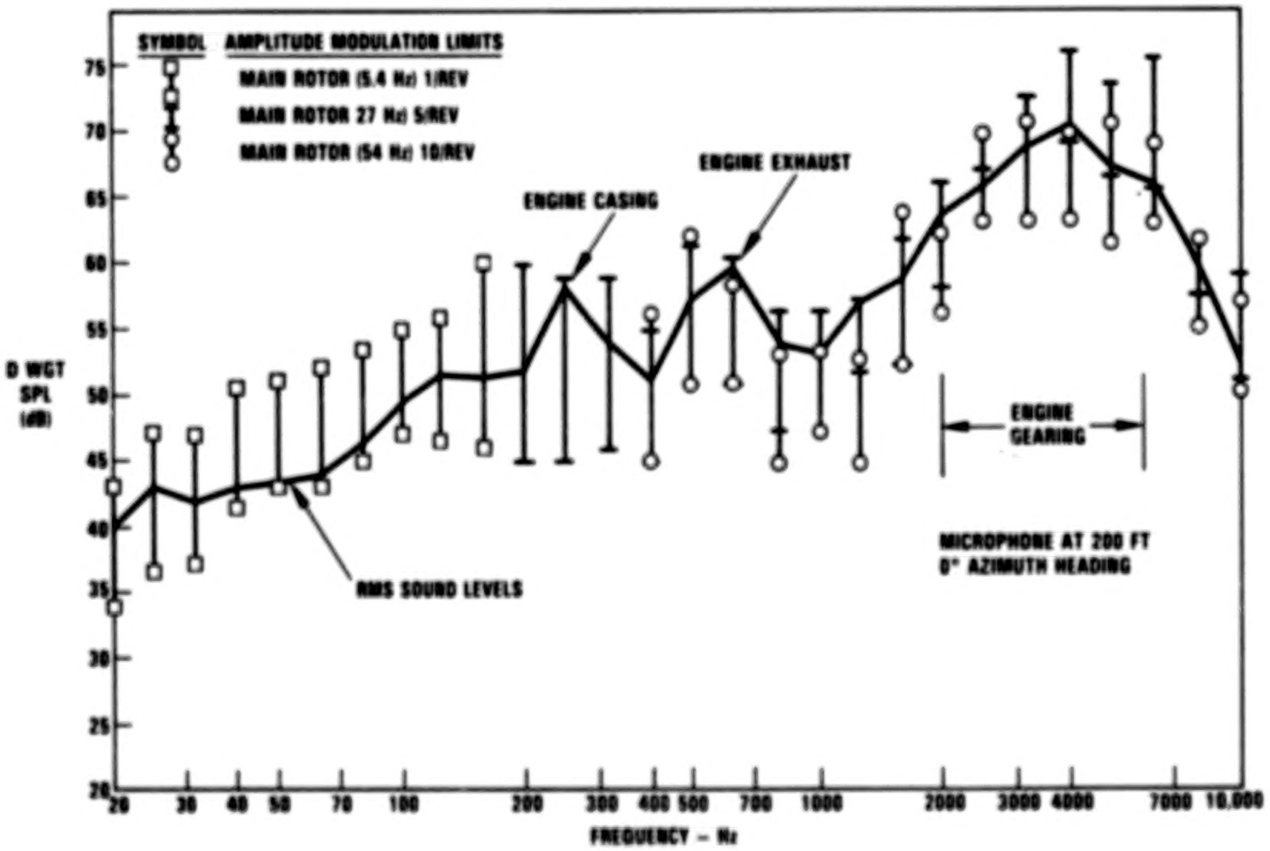


Figure 7

NOISE COMPARISON OF YO-3A AND THE QUIET ONE

An interesting part of the Quiet Helicopter program was the fly-off of the Quiet One helicopter against the quiet fixed wing airplane, the YO-3A, which was built for surveillance work in Viet Nam. Figure 8 shows the overhead noise spectrum of the two aircraft, both cruising at an altitude of 125 ft and forward speed of about 75 kt. The Quiet One is almost as quiet as the YO-3A except at higher frequencies. The flight tests also showed that the Quiet One was actually quieter than the YO-3A when judged on the basis of detection range. Both aircraft were extremely quiet. However, the YO-3A projected forward a higher frequency propeller noise which made it detectable almost two times farther away than the quiet helicopter.

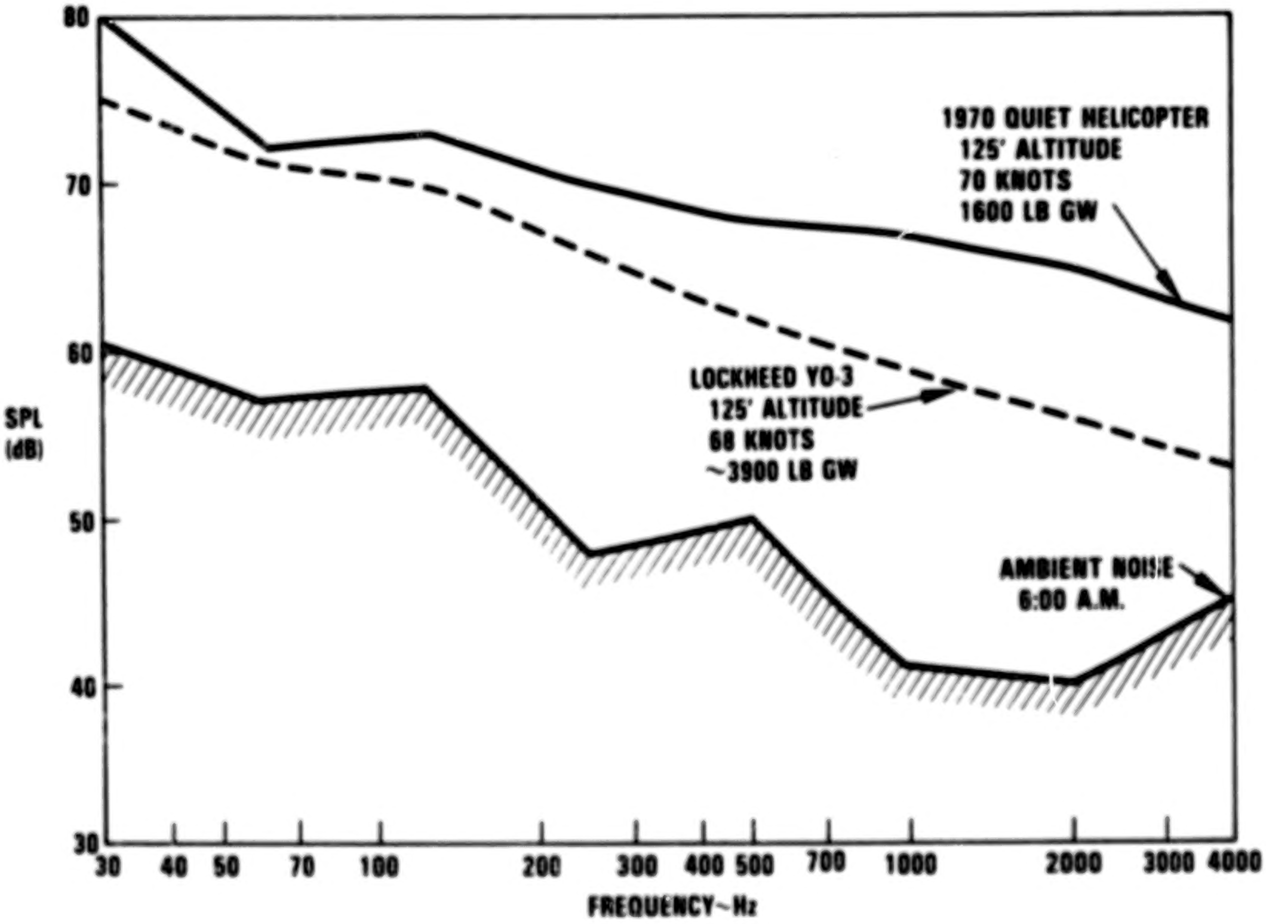
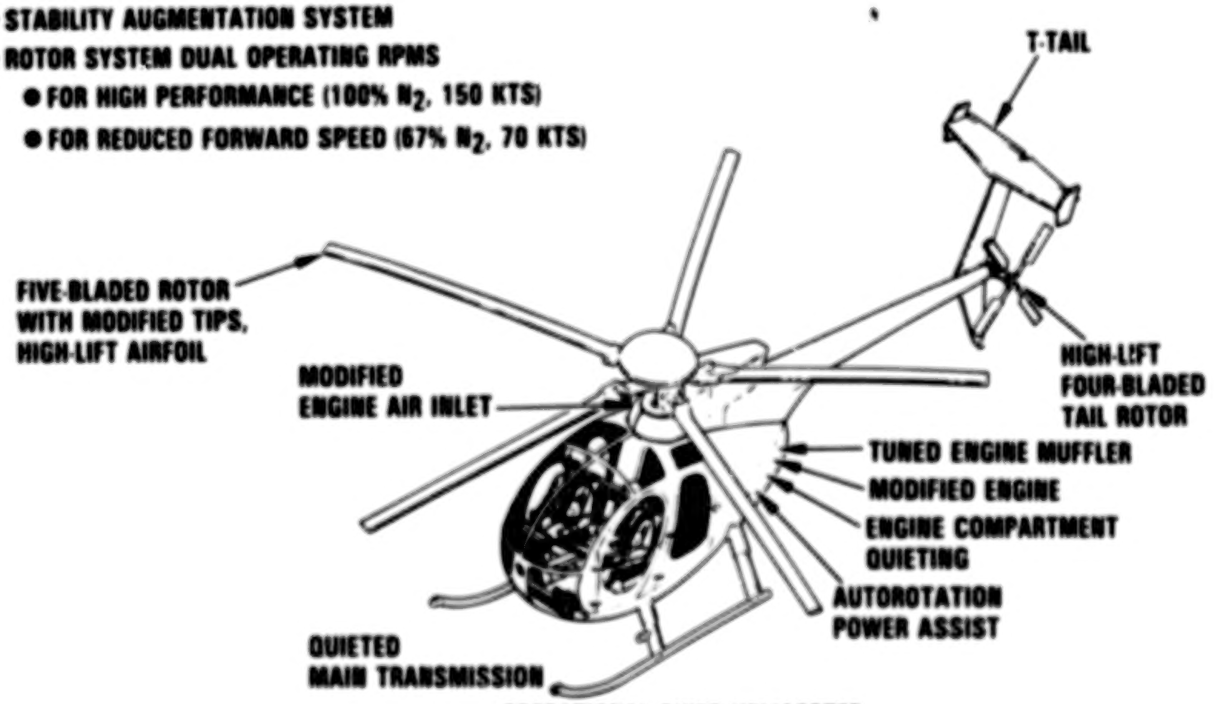


Figure 8

OPERATIONAL QUIET HELICOPTER

The reduced rpm flight mode and other quieting modifications of a quiet helicopter create some operational problems that must be considered in the trade-off studies when the value of quieting is determined. Some of the considerations are: development of a wide speed range engine governor and engine modifications, rotor dynamics over a wide rpm range, reduced cooling availability at low engine rpm, effect of center of gravity shift and increased weight, maintenance of proper frequency control of electrical power, autorotation close to ground at lower rpm, and reduced stability and control at low rpm. Based on the Quiet Helicopter Program, a quiet operationally feasible light observation helicopter, a derivative of OH-6A, was defined (fig. 9). As detailed in reference 3, it would be "The Quiet One" with an autorotation assist and stability augmentation system (SAS). Its rotor system would have a dual operating rpm provision, one for high performance (100% N_2, 3150 lbs gross weight and V_{ne} = 150 kts) flight mode and the other a low rpm (67% N_2) quiet mode at reduced gross weight (1600 lbs) and forward speed (70 kts).

- THE QUIET ONE WITH
  - AUTOROTATION ASSIST
  - STABILITY AUGMENTATION SYSTEM
  - ROTOR SYSTEM DUAL OPERATING RPMS
    - FOR HIGH PERFORMANCE (100% N_2, 150 KTS)
    - FOR REDUCED FORWARD SPEED (67% N_2, 70 KTS)


T-TAIL
FIVE-BLADED ROTOR WITH MODIFIED TIPS, HIGH-LIFT AIRFOIL
MODIFIED ENGINE AIR INLET
HIGH-LIFT FOUR-BLADED TAIL ROTOR
TUNED ENGINE MUFFLER
MODIFIED ENGINE
ENGINE COMPARTMENT QUIETING
AUTOROTATION POWER ASSIST
QUIETED MAIN TRANSMISSION

OPERATIONAL QUIET HELICOPTER

Figure 9

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COMPONENT NOISE SOURCE TESTS - TEST RIG

A special test rig (fig. 10) was designed and built to conduct component noise source tests for a light observation helicopter of the OH-6A type. The objective was to isolate the various noise sources (main rotor, tail rotor and engine) of the helicopter and quantitatively evaluate, penalties incurred as various techniques were used to reduce their noise levels. The test rig as shown in figure 10 contained a dynamometer for absorbing engine power and an exhaust silencing system for reducing engine noise. The test set-up allowed various components of the helicopter (main rotor, tail rotor and engine) to be run in simulated hover (6 ft skid height) and listened to individually or in any combination. The sound pressure levels were recorded 200 ft away at 30° left of aft centerline. More details of the test rig and the noise measurements are given in reference 4.

Figure 10

COMPONENT NOISE SOURCE TESTS - RESULTS

The special test rig provided noise data for individual noise sources which can be used in the validation of rotor noise prediction methods. Figure 11 shows the noise spectra in 1/3 octave bands of component sources (main rotor, tail rotor,engine) and of the complete aircraft (standard OH-6A at 2400 lbs, 103% N_2) in simulated hover. The effects of various quieting techniques used to reduce the noise of individual sources and weight penalties incurred in the process were obtained in the form of partial derivatives. It was concluded that for a light observation helicopter of the OH-6A type, a substantial decrease in external noise could be obtained by reducing the tail rotor tip speed with only a small penalty in lost payload. It was also shown that increasing the main rotor number of blades and reducing its tip speed produced only a small reduction in noise while incurring a significant weight penalty. This was probably due to the low main rotor tip speed of the OH-6A and could still be an effective noise reduction technique for a helicopter with higher tip speed.

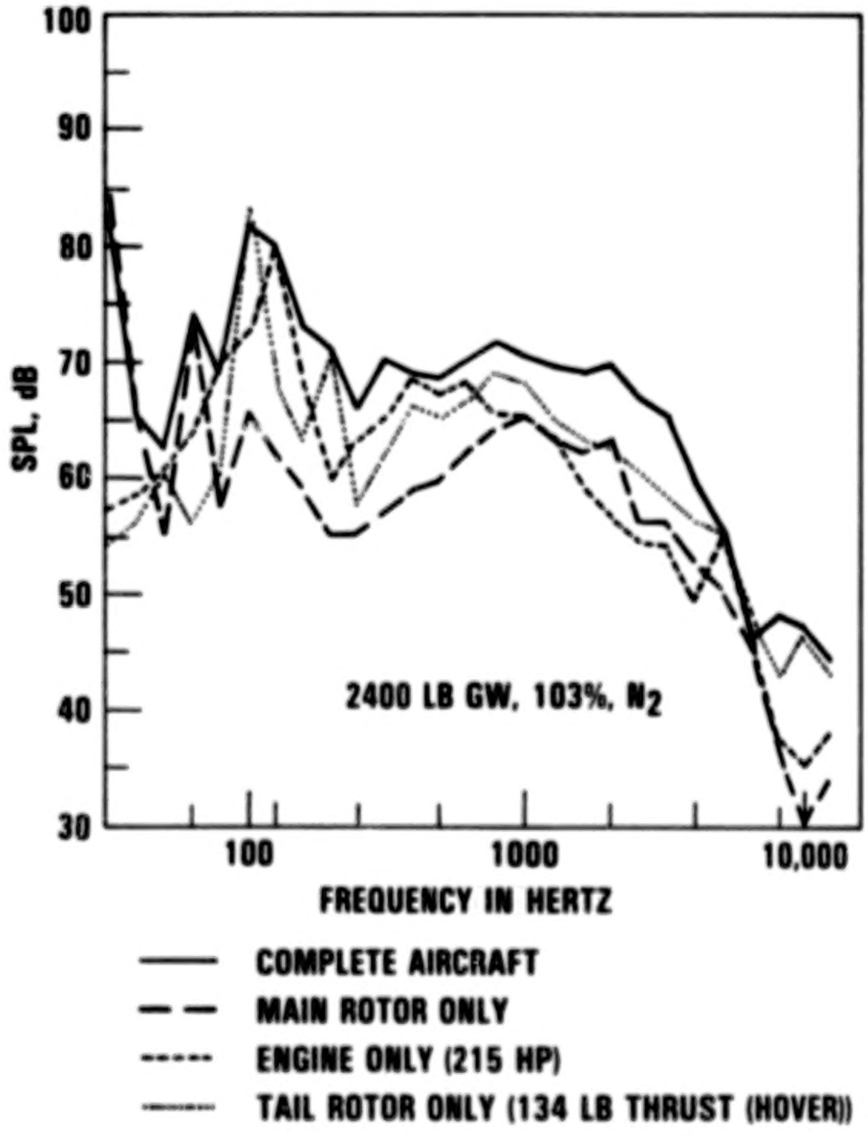

Figure 11

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EFFECT OF BLADE PHASING

As part of the component source tests, the effect of tail rotor blade stagger and phasing on its noise was determined. Figure 12 shows 1/3 octave band noise spectra of four different tail rotor configurations at approximately the same operating conditions. In the case of four-bladed tail rotors, three different configurations differing in the blade phasing (90°/90°, 60°/120° & 75°/105°) were tested. It was found that a blade phasing of 75°/105° was the best from the noise point of view. Four-bladed rotors with 60°/120° and 75°/105° phasing have noise spectra similar to that of a two-bladed rotor (see fig. 12). Compared to a two-bladed rotor, a four-bladed rotor with 75°/105° phasing showed 5 dBA noise reduction. It should be noted that the four-bladed tail rotor had twice the solidity of the two-bladed rotor.

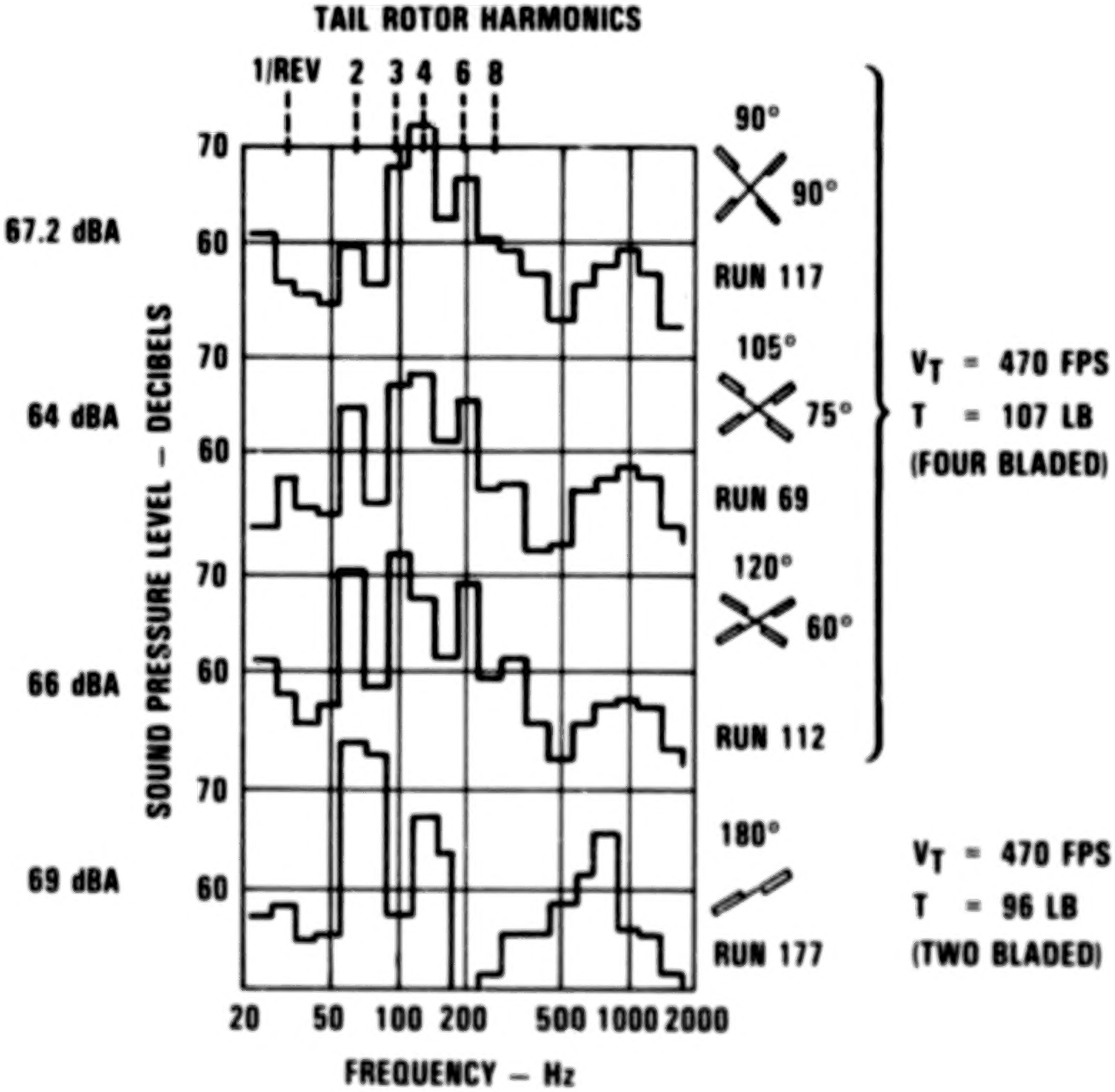

Figure 12

H500D - QUIET VERSION

The H500D is a derivative of the OH-6A. It has a five-bladed main rotor (tip speed 680 fps) and a two-bladed tail rotor (tip speed 704 fps) with a gross weight of 3000 lbs and V_{ne} of 152 kts. Based on the quiet helicopter experience, a quiet version of the H500D was developed. It has the same main rotor as that of a standard H500D, but with a four-bladed tail rotor (twice the solidity of the standard H500D) and blade phasing of 75°/105° operating at a reduced tip speed of 530 fps. The tail rotor modifications resulted in an insignificant increase in empty weight of 9.4 lbs. Noise flight tests of an H500D with standard and with quiet tail rotor were conducted in the cruise, max-power climb, and descent flight modes per ICAO procedures. Figure 13 shows the overhead noise spectra of the H500D and its quiet version in cruise flight at an altitude of 492 ft and a forward speed of 114 kts (.9 V_H). It shows that the quiet tail rotor resulted in noise reductions over a frequency range of 100 to 10,000 Hz. The quiet tail rotor reduced the noise of the H500D by 3 PNdB for the overhead position.

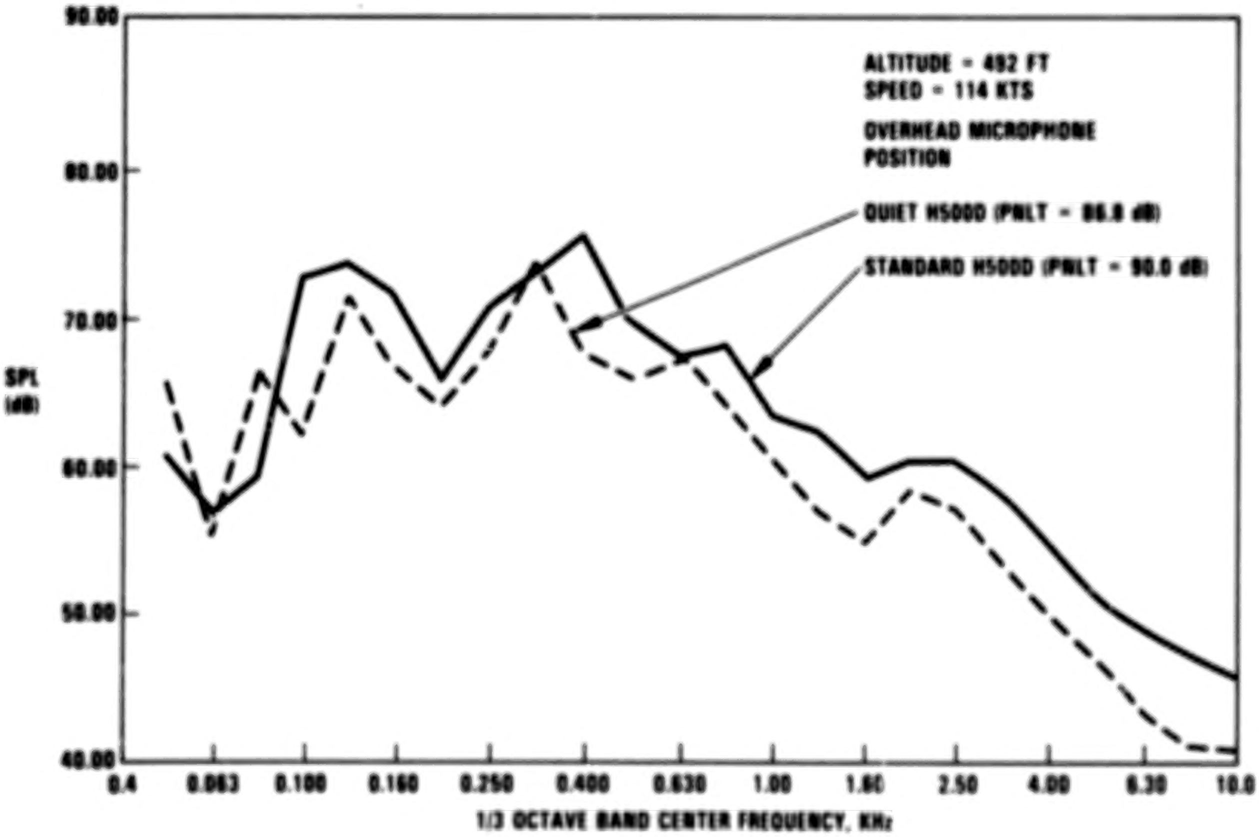

Figure 13

QUIET H500D FLIGHT TESTS - DESCENT

Figure 14 shows the 1/3 octave band spectra of the standard and quiet H500D helicopters for the descent flight condition. The data corresponds to a microphone position 550 ft forward of the helicopters. As shown in figure 14, the quiet H500D is slight more noisy (.8 PNdB) than the standard H500D for this microphone position. It suggests that the predominant noise source for this directivity is probably the main rotor blade-vortex interaction. In addition, the descent flight is a low power flight with low tail rotor thrusts and therefore the effect of the quiet tail rotor is not felt. However, for the total descent flight duration, the quiet H500D showed a marginal reduction of 1.0 EPNdB over that of the standard tail rotor.

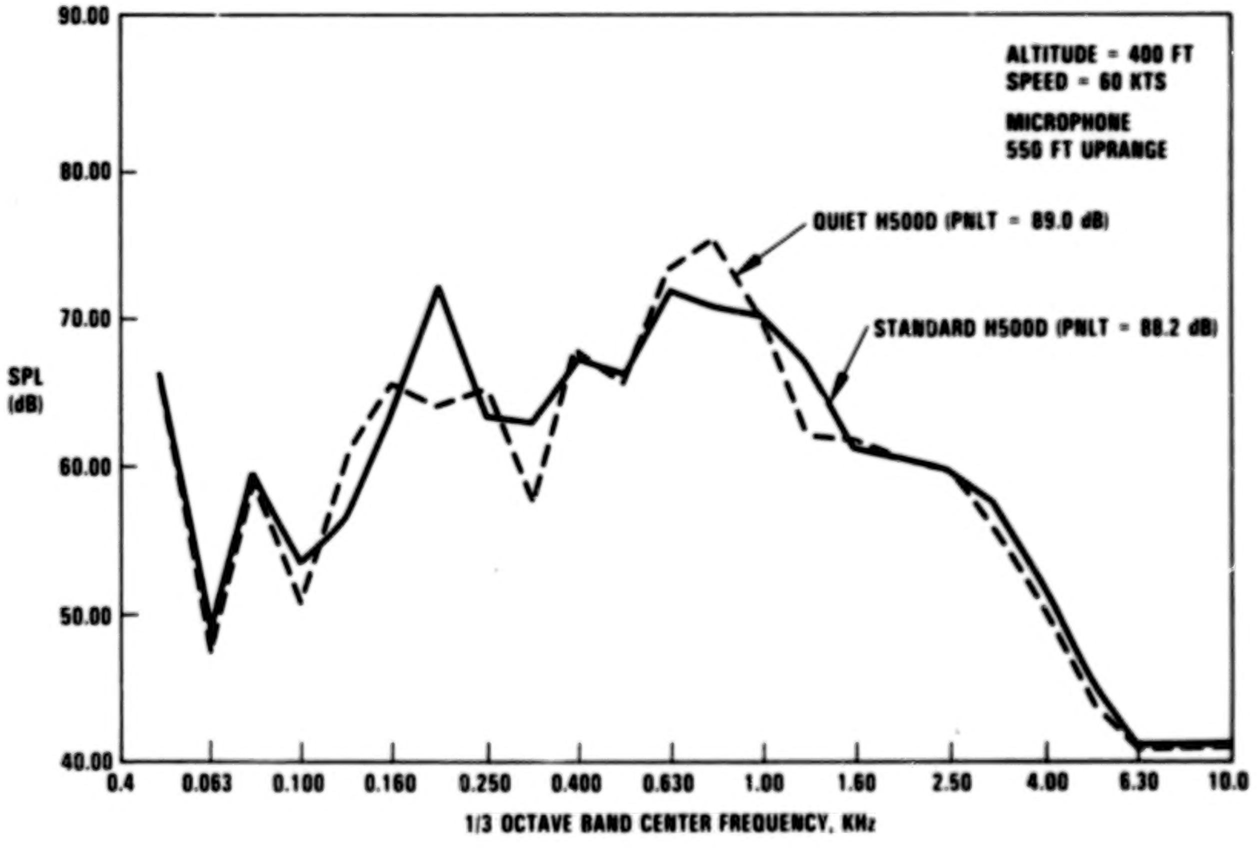

Figure 14

QUIET H500D FLIGHT TESTS - SUMMARY

The table below shows the noise levels of the H500D and its quiet version in three flight modes. The noise levels are given in EPNdB and are averages of the data obtained at three microphones laid out per the ICAO procedure. It is seen that the quiet H500D showed an average of about 2.8 EPNdB noise reduction. Thus, a significant quieting with insignificant weight or performance penalties was achieved with the quiet version of the H500D.

Flight Condition	Standard H500D EPNdB	Quiet H500D EPNdB	Difference EPNdB
Cruise at .9 V_H (115 kts)	87.7	83.8	-3.9
Approach (6° glideslope) at 60 kts	88.9	87.8	-1.1
Max power climb at 55 kts	88.7	85.3	-3.4

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300 CQ HELICOPTER

Figure 15 shows a 300CQ helicopter, a quiet version of 300C which is a light piston-engined helicopter with a gross weight of 2050 lbs. Quieting for this helicopter is provided only in the cruise flight mode (between 40 and 70 mph) at a reduced gross weight of 1925 lbs and a minimum operating altitude of 500 ft. The quieting features consist of reduced rotor rpm (by 10%) and the use of dual exhaust mufflers. The quiet operation results in a noise reduction of about 4 dBA in 500 ft flyover at a payload loss of about 20%.

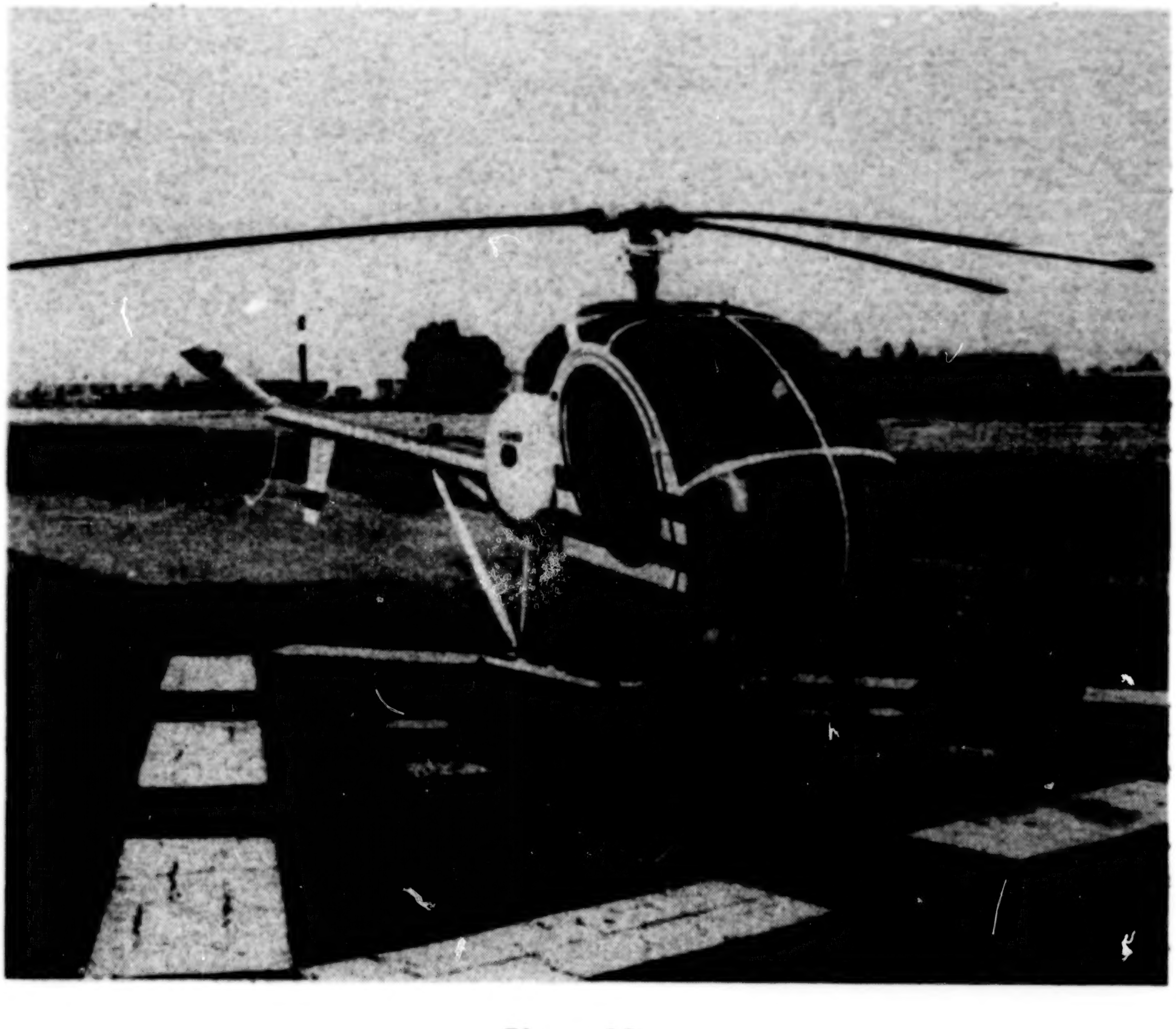

Figure 15

HHI NOISE REDUCTION EXPERIENCE - SUMMARY

Noise reduction experience at Hughes Helicopters, Inc. was primarily confined to derivatives of aircraft originally designed for a performance mission without any regard to noise considerations. It was also mostly limited to light helicopters whose noise signature is dominated by their tail rotors. It was primarily hardware oriented. Well-known noise reduction techniques such as reduction of rotor speeds with an accompanying increase in solidity to maintain performance, engine noise reduction with the use of exhaust mufflers,and acoustic blanketing of transmission and engine compartment were used. The concept of blade phasing as a means of reducing tail rotor noise was also used.

- Primarily on derivatives
- Limited to light helicopters
- Hardware oriented
- Use of established noise reduction techniques
    - Reduction, rotor tip speed
    - Accompanying solidity increase
    - Engine exhaust mufflers
- Tail rotor blade phasing

It was found that engine noise (exhaust noise), power train noise and airframe noise will become important at low rotor tip speeds and means must be found to reduce these noise sources if further noise reductions are desired. The use of a special test rig was very helpful in isolating the various noise sources and arriving at the penalties (performance or payload) involved in quieting them. It was possible to achieve significant noise reductions for the light helicopter at HHI with minimum performance or weight penalties because of the dominance of a single noise source (the tail rotor).

- Importance of engine noise at low rotor speeds
- Development and use of a special test rig
- Able to achieve quieting with minimum penalties due to single source dominance

NOISE REDUCTION - DERIVATIVE (EXISTING) AIRCRAFT - PREFERRED APPROACH

Based on the noise reduction experience at HHI, the following approach is preferred for acoustic improvement of existing aircraft designs. First, using test and prediction programs, identify the major sources in the noise signature of the original aircraft for the important flight conditions. If the tail rotor is the dominant noise source, it is relatively easier to achieve quieting with minimum performance and weight penalties. Some of the noise reduction techniques that should be considered are reduction in tip speed, accompanying increase in solidity (usually increase in blade number), tail rotor blade phasing, and change in direction of rotation. However, if the main rotor is the dominant source, it is necessary to explore the benefit of the new technology blades before considering the higher design impact changes. First determine the benefit of blade tip shape changes and high lift airfoils. Thinning the blade tips is helpful especially from the point of view of high speed thickness noise. Next, define the tip speed and solidity changes necessary to meet the noise goal. If the engine is one of the primary sources of noise (a rarity in present day helicopters), efforts should be concentrated on the development and use of an efficient exhaust muffler. It is of course necessary to continually evaluate the effect of various noise reduction techniques on the performance of the aircraft.

- Identify noise sources, test, predictions
- Tail rotor noise dominant - quieting easier
- Main rotor noise significant
    - Tip speed/solidity changes
    - Tip shapes
    - Airfoils
    - Thinning the tips
- Engine noise significant (rare today)
    - Exhaust muffler
- Use of relatively simple noise prediction programs

NOISE REDUCTION - NEW AIRCRAFT - PREFERRED APPROACH

It is necessary to have validated noise prediction programs to design a totally new rotor system or helicopter for noise. First, relatively simple prediction programs should be used to arrive at the gross rotor parameters which satisfy both performance and noise requirements for the flight conditions of interest. Then it would be necessary to use more sophisticated prediction programs for both noise and performance to arrive at the detailed rotor parameters. Trade-off studies will be required and it will be necessary to iterate between noise and performance to help select the detailed rotor parameters. The effects of the design on weight should also be evaluated. Some of the important rotor parameters from the point of view of noise are tip speed, blade number, blade tip shape, airfoils, blade tip thickness, and (for tail rotors) blade phasing, and direction of rotation.

- Need of validated prediction programs
- Performance and noise trade-offs
  - Use of simple prediction programs
  - Selection of gross rotor parameters to satisfy performance and noise
- Detailed rotor system design for noise
  - Use of more sophisticated noise prediction programs
  - Effects on performance
  - Iterations if necessary

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BOEING VERTOL NOISE REDUCTION EXPERIENCE

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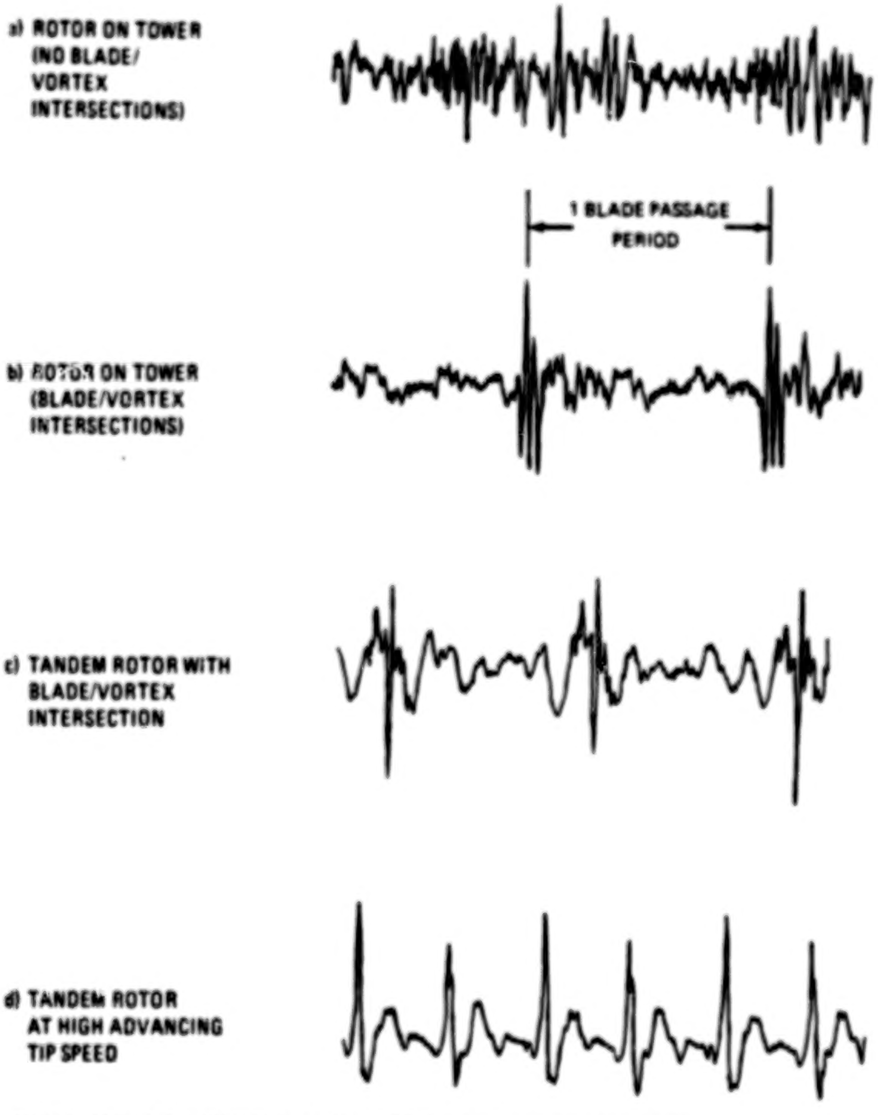
INTRODUCTION

Boeing Vertol Model 234, CH-47D and 347 tandem rotor helicopters have demonstrated noise levels that comply with the overall requirements of the noise standards recently considered by the FAA and similar standards currently being considered by ICAO member nations. These helicopters achieved these noise levels because they are, by design, free of impulsive noise in level flight and climb. Avoiding an impulsive noise signature was achieved by developing and applying configuration design criteria that eliminated blade vortex intersections and recognized advancing blade tip noise limitations in high speed forward flight. Airfoil design, rotor separation and rotor tipspeed selection all contributed to the noise control demonstrated in the Model 234, CH-47D and 347 helicopters.

Although the Model 234 and the CH-47D meet the overall requirements of the ICAO/FAA noise standards, compliance with the approach element of the standards is marginal at best. The approach problem is not appreciably alleviated by reduced tipspeed and airfoil design. Since it is a single rotor phenomenon, it is not alleviated by rotor separation in the tandem configuration. Tip shapes may provide the answer, but to date this approach has not led to a satisfactory solution.

HELICOPTER ROTOR NOISE SIGNATURES

Helicopter rotor sound pressure time histories are shown in Figure 1. Data recorded during a single rotor whirl tower test in near zero winds are shown in Figures 1a and 1b. The sound pressure time history in Figure 1a displays no impulsive noise content at blade passage frequency. Smoke released into a blade tip vortex during this test showed that there were no blade intersections with tip vortices. The impulsive sound pressure time histories at blade passage frequency shown in Figure 1b resulted from blade intersections with tip vortices which were corroborated by smoke visualization of the tip vortices. The sound pressure time history of a tandem rotor helicopter flyby is shown in Figure 1c. The impulsive signature resulted from the intersection of rear rotor blades with tip vortices generated by the forward rotor. The noise signature of a high speed flyby is shown in Figure 1d. The advancing blade tip speed is well in excess of the blade drag divergence Mach number which caused the impulsive pressure wave to emanate from the advancing tips of both the forward and aft rotor blades, thereby producing impulses at twice the single rotor blade passage frequency.



NOTE: AMPLITUDES HAVE BEEN APPROXIMATELY NORMALIZED.

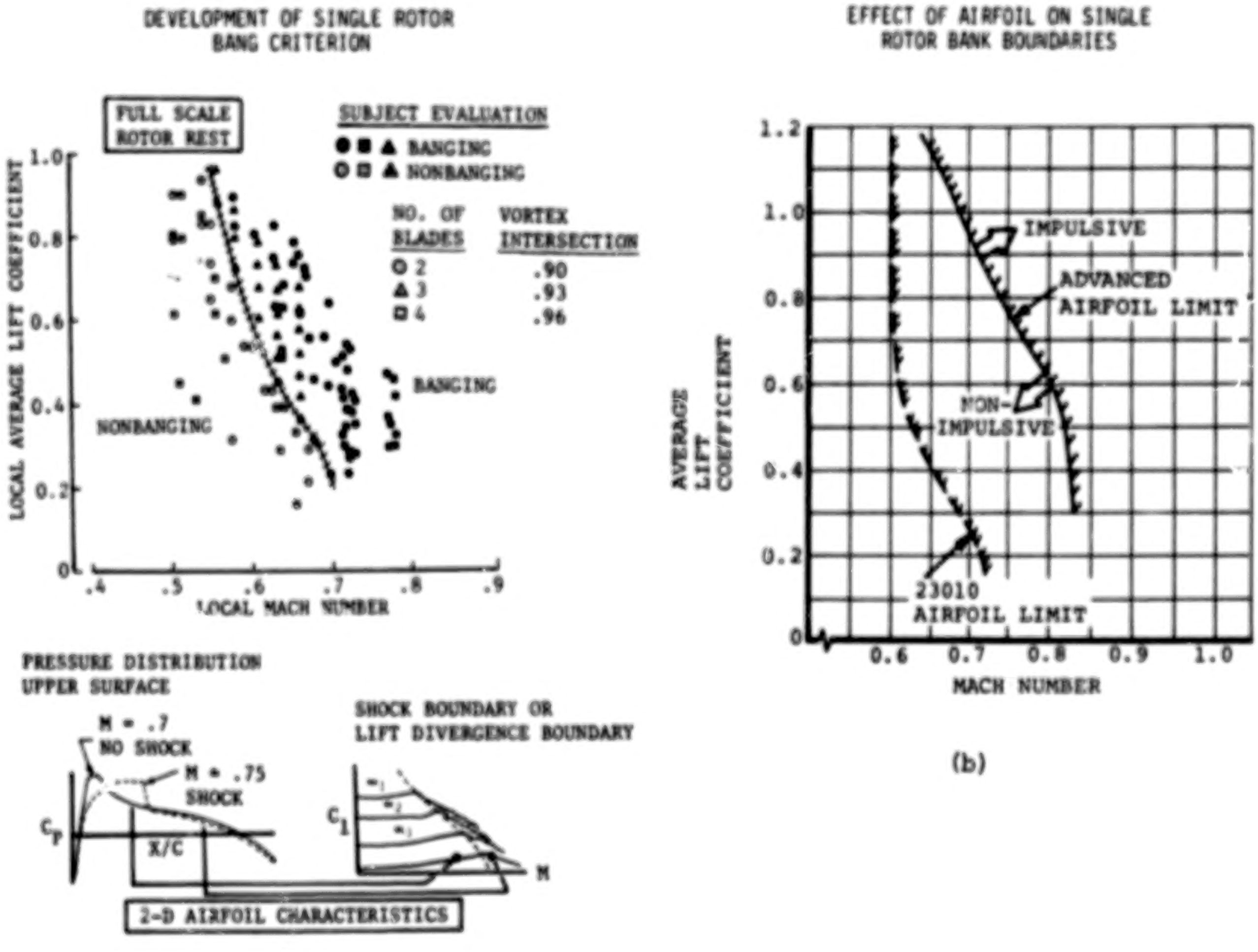
Figure 1

CRITERIA FOR IMPULSIVE NOISE AVOIDANCE - SINGLE ROTOR

Successful noise control for tandem rotor helicopters relies on design criteria that eliminate or minimize conditions that produce impulsive noise. Without impulsive noise, the remaining noise levels can meet currently envisioned noise regulations. The discussion following presents the empirical noise control design criteria used at Boeing and illustrates the application of these criteria to derivative models of the CH-47A helicopter which produced the successively lower noise levels achieved in the CH-47B, CH-47C and, finally, the CH-47D, Model 234 and Model 347.

Boeing full scale and model scale experiments with hovering rotors have shown that an impulsive noise is generated at critical combinations of lift and local Mach number when blade vortex intersections occur. A design criterion has been developed (Figure 2a) based on airfoil shock boundaries that precludes generation of impulsive noise from a single hovering rotor, and has demonstrated good correlation with rotor tower test data.

Design of advanced airfoils takes this boundary into consideration. The boundary has been considerably extended (Figure 2b) by the advanced airfoils presently incorporated in the CH-47D and Model 234.



(a)

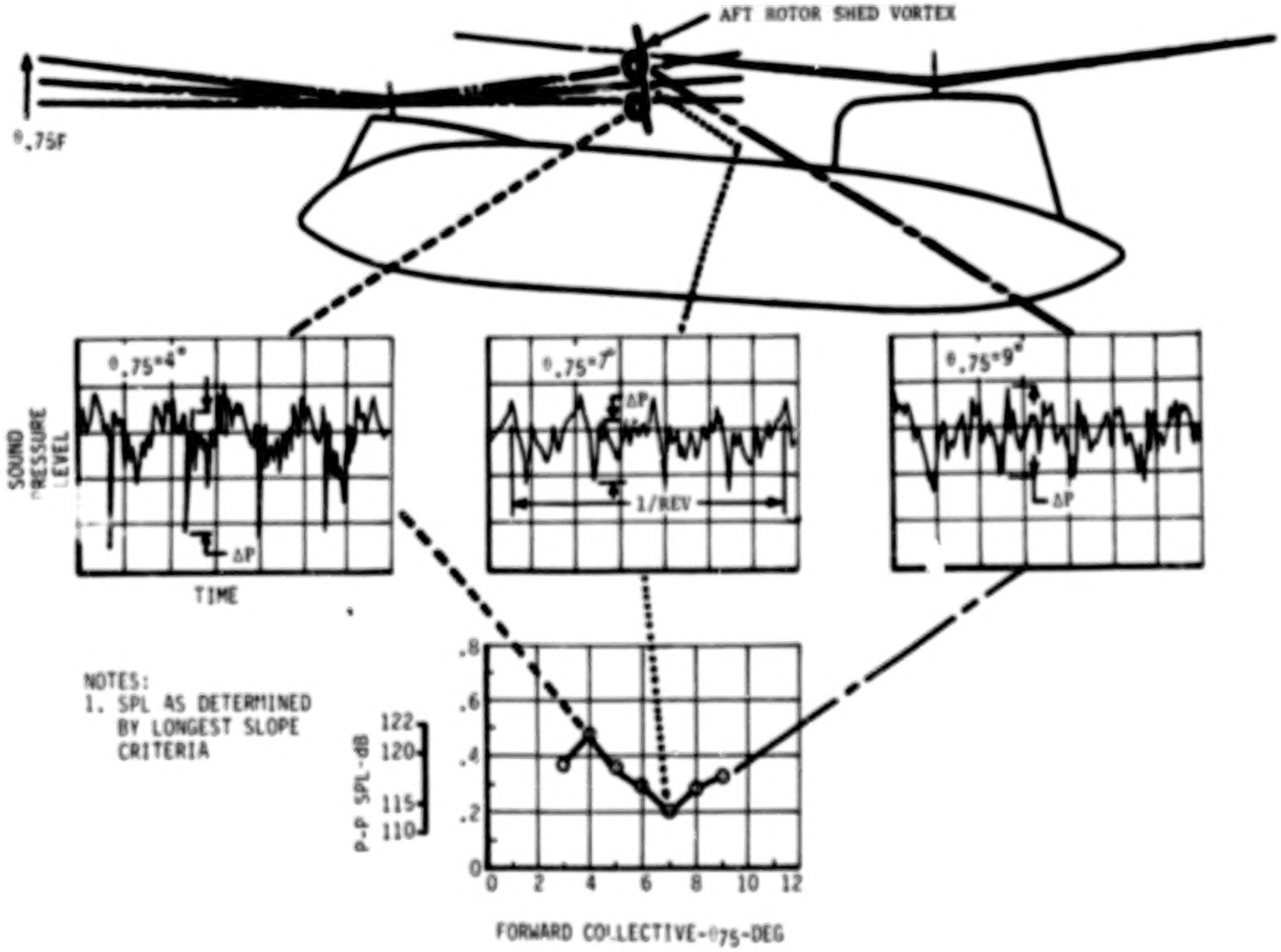
(b)

Figure 2

CRITERIA FOR IMPULSIVE NOISE AVOIDANCE - TANDEM ROTOR

Impulsive noise associated with overlapped tandem rotor configurations results from rotor-rotor interaction effects. A vortex trailed by a blade on one rotor is intersected by a blade on the other rotor and the resulting velocity disturbance causes impulsive pressures on the intersecting blade. Figure 3 illustrates the manner in which blade-vortex separation is affected by flapping (in this case resulting from application of collective pitch). The same effect can be obtained from differential longitudinal cyclic pitch, which varies longitudinal flapping and also varies the vertical separation of blade and vortex.

In-flight vortex visualization was used to verify blade-to-vortex separation at several airspeeds and longitudinal cyclic trim conditions. The conditions illustrated in Figure 4 show the relation between peak sound pressure level, separation between the blade and vortex, and the subjective rating of impulsiveness. A method has been developed using a simplified rigid wake analysis for achieving the blade vortex separation required by Figure 4 in order to avoid banging, and this has proved successful when applied to more recent tandem rotor helicopter design.



NOTES:
1. SPL AS DETERMINED BY LONGEST SLOPE CRITERIA

Figure 3

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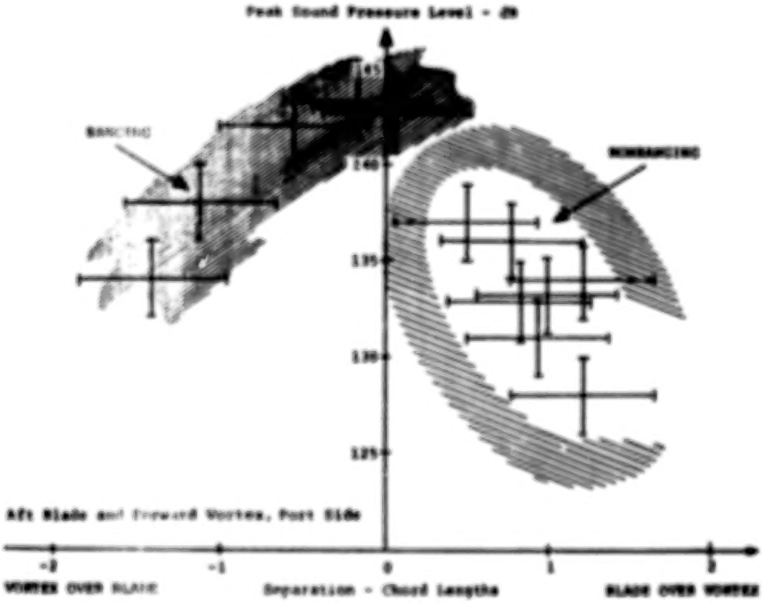
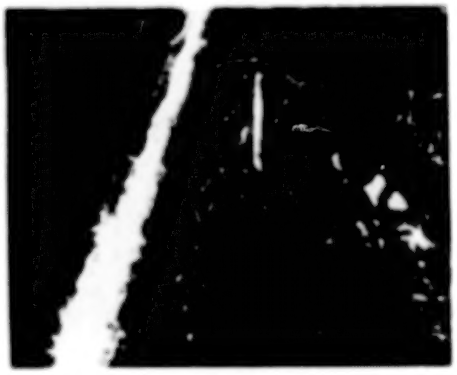
CH-46A
0, 40, 75, 120 kt
Ref: Study of Rotor Blade Tip Vortex Geometry for Noise and Airfoil Applications, Naval Air Systems Command, June 1969.

Vortex Seen from Camera
on Test Aircraft.





Vortex Seen from Camera
on Chase Aircraft.

Figure 4

CRITERIA FOR IMPULSIVE NOISE AVOIDANCE - HIGH SPEED

Another impulsive noise source is related to drag divergence occurring on the advancing blade tip. This impulsive noise occurs somewhat above the drag divergence Mach number, M_{DD}, for the airfoil section.

Figure 5 shows the delay in onset of the buildup of high speed impulsive noise that can be derived from thin airfoils as demonstrated by both model rotor wind tunnel and full scale flight test. The flight test data show no real advantage of thin tips below about 0.88 Mach number or above 0.94 Mach number, but a significant benefit can be obtained between these two limits which corresponds to the high forward speed range of new generation helicopters. These data were recorded on the Boeing Vertol Model 347 for conditions where no blade vortex interactions were occurring.

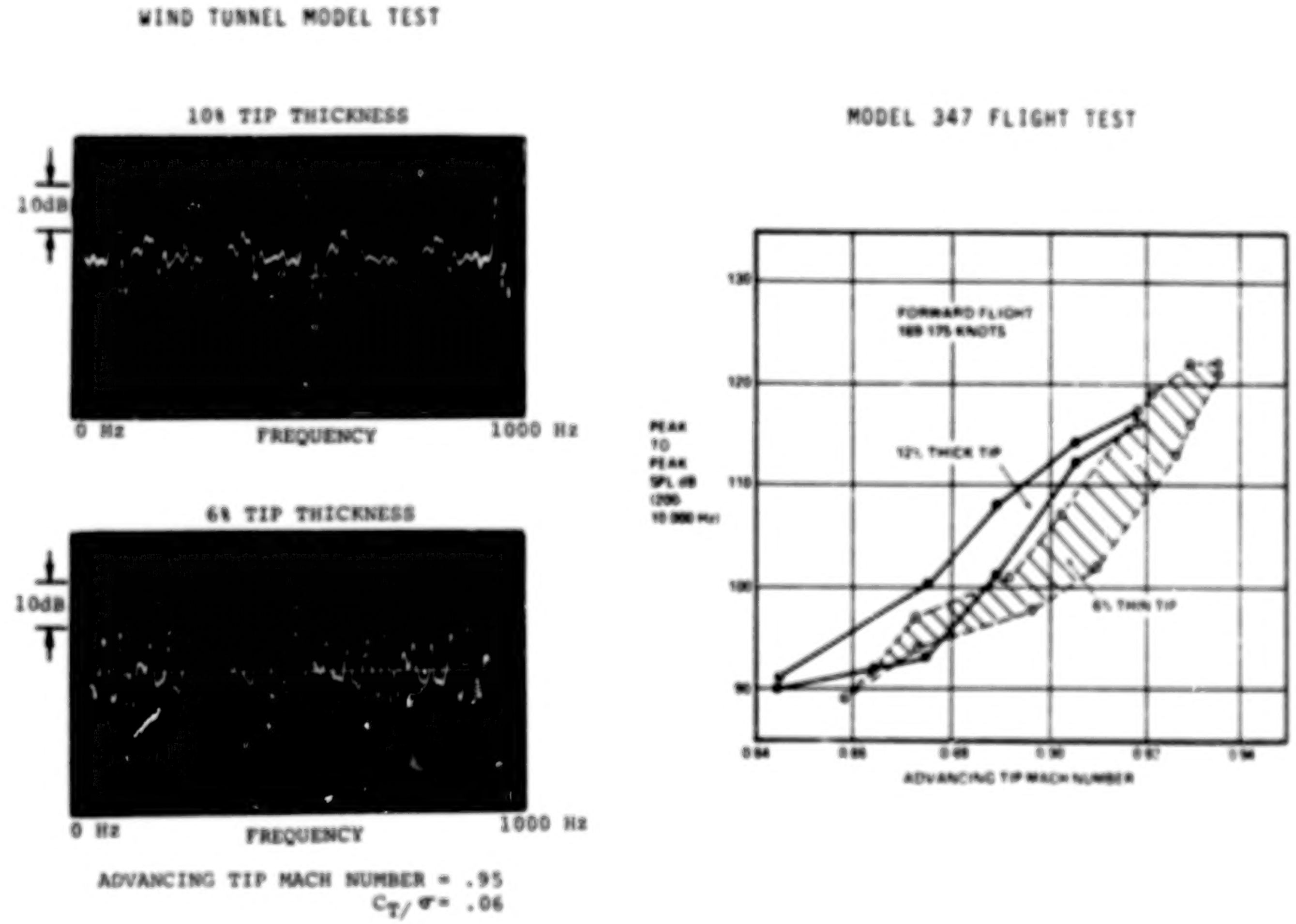


Figure 5

NOISE REDUCTION EXPERIENCE - CH-47 DERIVATIVES

The CH-47 helicopter has undergone four model changes during its service life which now is in excess of 20 years. Primary changes have been associated with powerplant and drive system improvements resulting in an increase in payload, speed and range. Maximum gross weights increased from 33,000 pounds to 50,000 pounds, and maximum speed has increased from 130 knots to 155 knots. During this time, the flyover noise of the CH-47 has been reduced by 17 EPNdB by substantially reducing or eliminating the impulsive components of rotor noise (see Figure 6). A reduction of 5 EPNdB from the CH-47A to CH-47B model resulted from a reduction in blade-vortex interactions between rotors by modifying the longitudinal cyclic trim schedule. This reduction in noise level was limited due to resulting increases in rotor shaft bending loads. Flyover noise of the CH-47C was reduced an additional 3 EPNdB by further modification of the cyclic trim schedule as permitted by a stronger rotor shaft. An additional 9 EPNdB reduction was achieved on the CH-47D model as a result of a reduction in rotor speed and a new wide chord rotor blade with an advanced airfoil and thinner blade tips. This design is similar to the Boeing Vertol Model 234. These reductions are maintained at substantial increases in gross weight. It is also believed that the less 'peaky' characteristics of the chordwise pressure distribution of the VR-7/8 airfoil (Figure 7) as compared with the Vertol 23010 contributed 5 EPNdB greater noise reduction than predicted. Criteria for this characteristic have yet to be developed.

FLYOVER NOISE LEVELS - BOEING VERTOL CH-47 DERIVATIVES

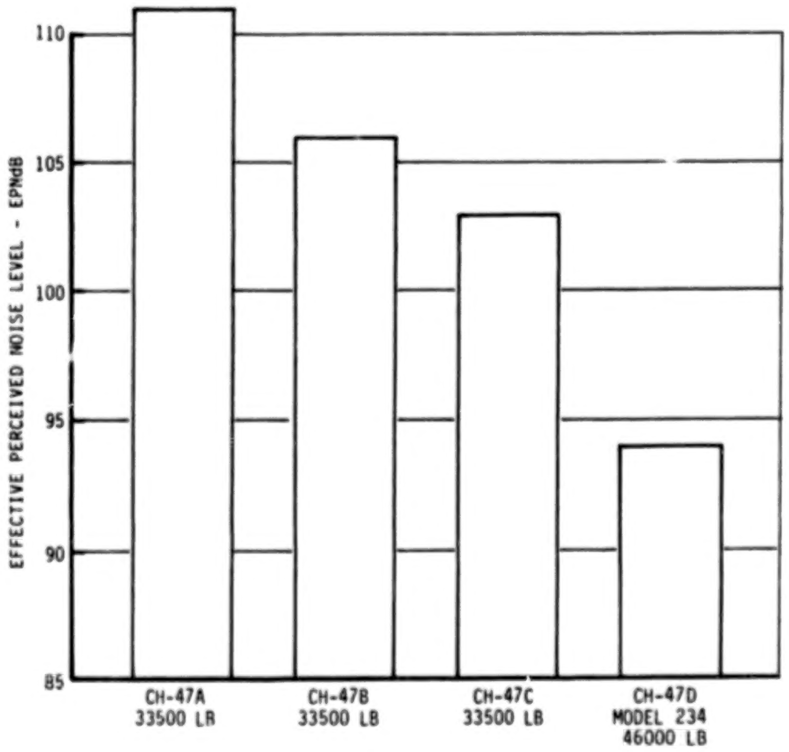


Figure 6

c-2

AIRFOIL PRESSURE DISTRIBUTIONS

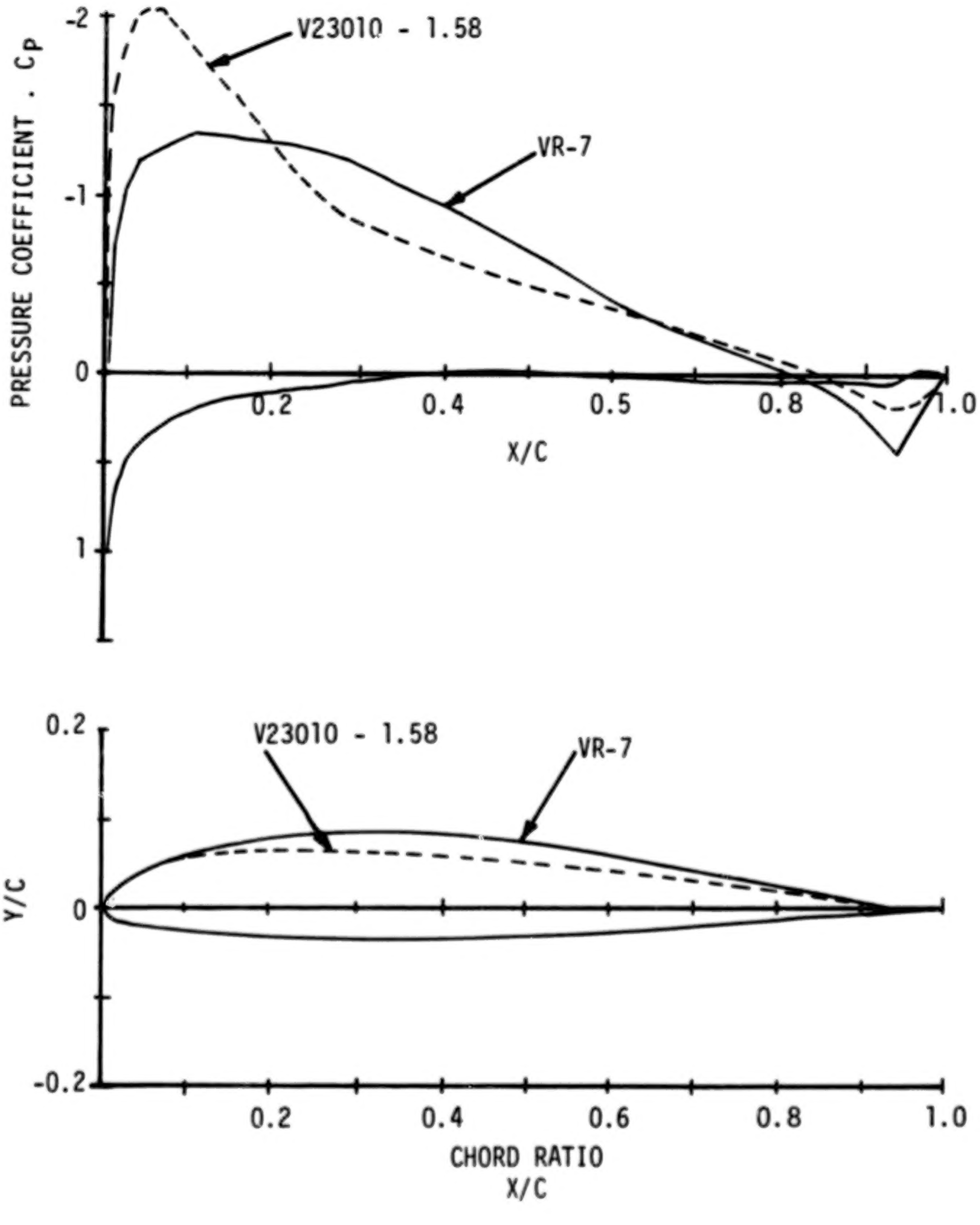


Figure 7

NOISE REDUCTION EXPERIENCE - MODEL 347

Low rotor noise levels were established as one of the design criteria for the Boeing Vertol Model 347 advanced technology demonstrator which underwent flight testing in 1970-1972. In order to reduce flyover noise levels from the CH-47 series, several modifications were made. First, the aft pylon was raised 30 inches and the fuselage was extended 110 inches to increase rotor separation, both vertically and horizontally. Second, the tip speed of the rotors was reduced from 770 to 691 ft/sec (while the number of blades was raised from 3 to 4) and, third, cambered airfoils incorporating thin tips replaced the conventional metal blade configuration. The Model 347 displayed no impulsive noise in flyover (Figure 8) as well as substantially reduced rotational noise levels as shown in Figure 9. Flyover EPNL was reduced to 90 EPNdB, a reduction of 21 EPNdB from the CH-47A (Figure 10).

WAVEFORM COMPARISON
FLYOVER NOISE

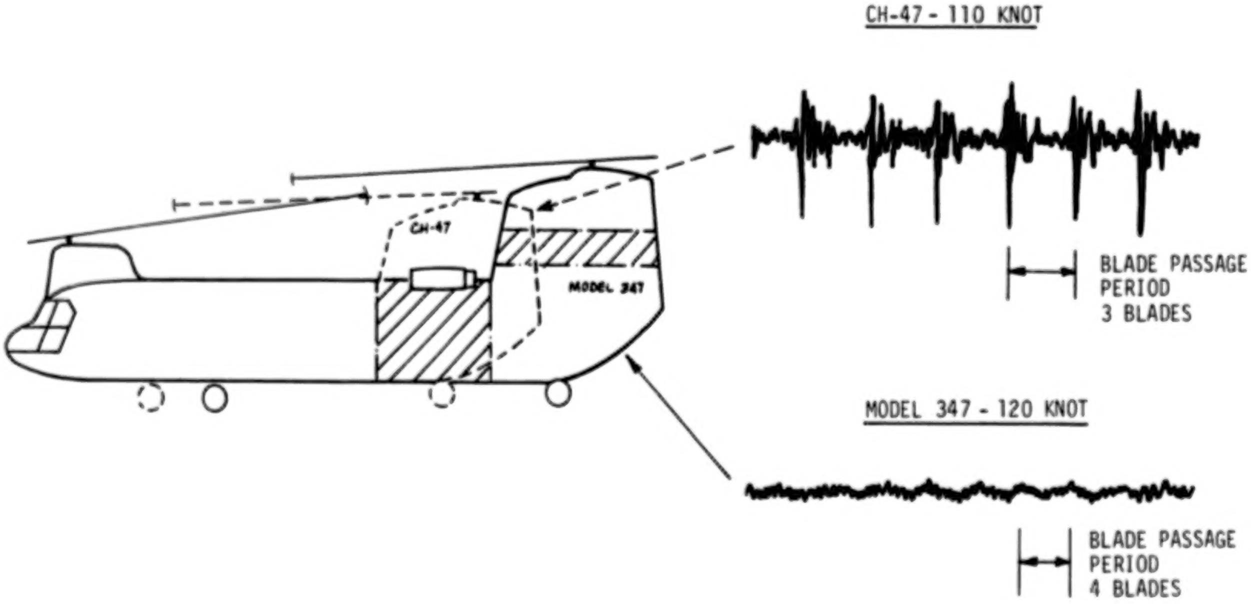

CH-47 - 110 KNOT
CH-47
MODEL 347
BLADE PASSAGE PERIOD 3 BLADES
MODEL 347 - 120 KNOT
BLADE PASSAGE PERIOD 4 BLADES

Figure 8

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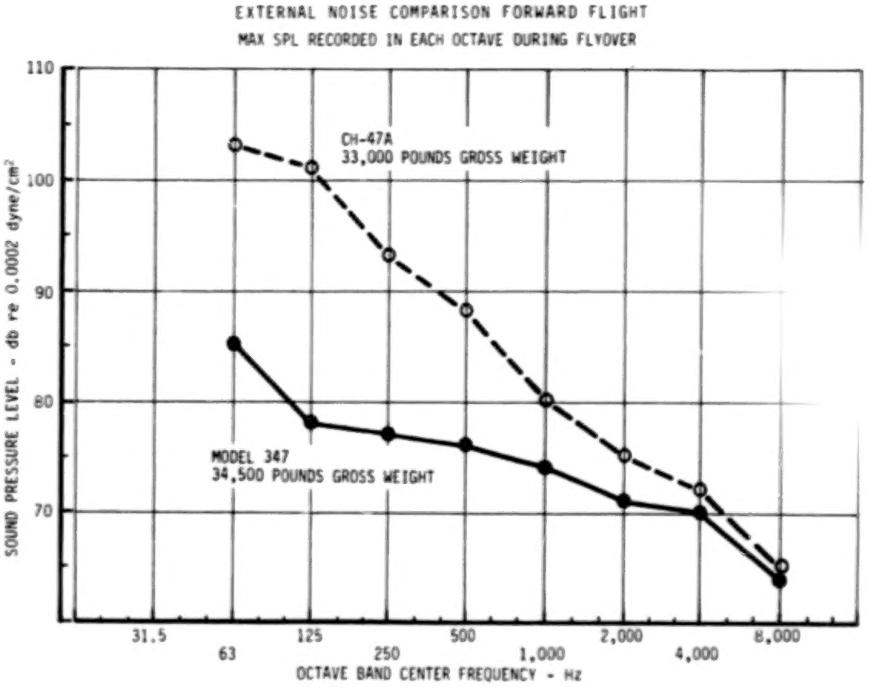

Figure 9

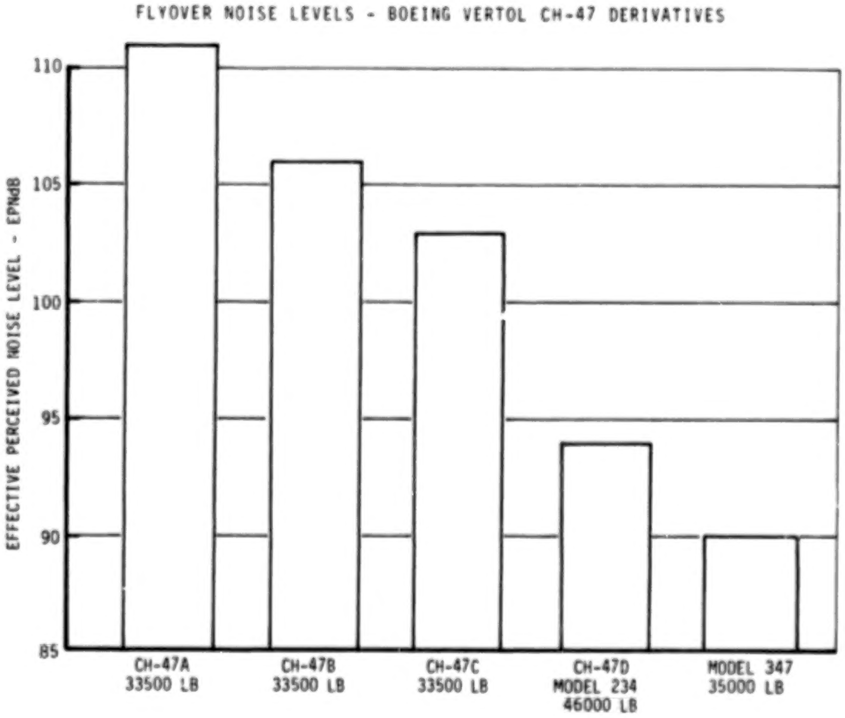

Figure 10

AREAS OF CONCERN

There are at least four areas of noise research which remain major concerns at present. One of these is the inability to eliminate impulsive noise during descent conditions. This noise mechanism is not well understood, and criteria for avoidance of impulsive noise during descent have yet to be developed. A second area of concern is noise at high forward speeds (200 knots), a flight regime that is emerging with the developments of recent technology advances and one which must be adequately addressed if the advantages of higher speeds are to be exploited. A third major area of concern is the inability to predict noise levels for all flight conditions for new designs with the confidence that is required to comply with noise rules. Underprediction leads to failure to comply with noise standards and the resulting costly modifications required to change the configuration after flight testing begins. Overprediction on the other hand results in inefficient rotorcraft and the associated economic impact on both manufacturer and operator. The fourth major concern is that current ICAO noise limits may unduly inhibit the development of large helicopters.

AREAS OF CONCERN

- REDUCTION OF IMPULSE NOISE IN DESCENT
- REDUCTION OF IMPULSE NOISE AT HIGH FORWARD SPEED (200 KNOTS)
- PREDICTION ACCURACY
    - FAILURE TO COMPLY WITH NOISE RULES
    - IMPACT ON AIRCRAFT DESIGN AND ECONOMICS
- THE CURRENTLY PROPOSED LIMITS OF THE ICAO STANDARD FOR VERY LARGE HELICOPTERS (80,000 TO 176,000 POUNDS GROSS WEIGHT)

Figure 11

N84 29668

ROTARY WING AERODYNAMICALLY GENERATED NOISE

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Moffett Field, California

WHY REDUCE NOISE?

The Army's concern for rotorcraft noise reduction was initiated in the early sixties and has become stronger over the years. Most of the military noise problems are parallel to those of the civil operator; community annoyance or acceptance and concern for hearing damage to the crew and passengers are in competition with the losses of performance and increased weight required for noise reduction. These problems have become more critical as the weight and power of helicopters have increased over the years. The Army has a different problem with acoustic detection and identification. The detection and identification capability has increased markedly with modern electronic techniques.

- **DETECTION**
- **DESIGN IMPLICATIONS**
- **COMMUNITY ACCEPTANCE**
- **HEARING DAMAGE**

US ARMY ACOUSTICS PROGRAMS, VIETNAM ERA

The government in general and the Army in particular have devoted considerable resources to rotorcraft noise reduction. The Quiet Helicopter Program, funded by A.R.P.A. and supported by the Army, was conducted to determine the technology available for noise reduction in the late sixties. Considerable reduction in external noise was obtained but costs in performance and weight were also significant. The QT-3 and limited production YO-3A quiet aircraft were also developed late in the period of the Vietnam Era.

The Army's research in rotorcraft acoustics was started in this period with the establishment of what is now the Research and Technology Laboratories (AVRADCOM). ARO-D initiated research programs in rotorcraft acoustics in 1969. The industry also became concerned with the noise problem and has developed and maintained a continual IR&D noise reduction effort. The U.S. industry put over $3 million into helicopter noise research and development in the period from 1960 to 1970.

US ARMY ACOUSTICS PROGRAMS, POST VIETNAM ERA

The post Vietnam Era shows a continued Army interest in rotorcraft noise reduction. Research had began to isolate and identify the technical voids and by the mid seventies the first in-flight full scale helicopter noise measurements were achieved. The Aeromechanics Laboratory developed the first anechoic hover facility and the Army included noise specifications in the requirements for the UTTAS and AAH development programs in the mid 70's. Full scale in-flight and model rotor experimental "stationary" data provided the measurement tool required to check scaling and prediction methods. More important, this high quality data assisted in the diagnosis and understanding of the physics of the problem. This process is essential to allow the technology to develop unconstrained.

The U.S. industry has also greatly increased its rotorcraft noise research and development effort. The total U.S. industrial effort is now in excess of twenty man-years per year.

PAST EXPERIMENTAL ACCOMPLISHMENTS

The Aeromechanics Laboratory aerodynamically generated noise research program has been directed toward building a technology foundation, concentrating on the most offensive of the noise sources, high speed thickness noise, that is dominant at high forward speed and is radiated forward in the plane of the rotor. This source is of importance on all modern helicopters and the fundamental cause and effect is independent of the number of blades and hub design. Initial experimental programs were directed toward understanding the full scale (flight) problem and developing tools and techniques for noise research.

- **IN-FLIGHT FAR FIELD ACOUSTIC MEASUREMENT TECHNIQUE**

UH-1H	**UTTAS**	**AH-1S**
AH-1G	**AAH**	**ETC.**

- **ANECHOIC HOVER CHAMBER (WORLD'S FIRST)**
- **HIGH SPEED SHADOWGRAPH AND SCHLIEREN ON MODEL ROTORS IN FORWARD FLIGHT**
- **GROUND BASED MEASUREMENTS ON UTTAS AND AAH**

More recently the experimental program has shifted more towards providing data to verify theoretical analysis and assuring the validity of model rotor acoustic scaling for both high speed thickness and blade vortex interaction noise.

- **MODEL SCALE/FULL SCALE ACOUSTIC TESTS WITH UH-1H IN 7 x 10**
- **MODEL SCALE/FULL SCALE ACOUSTIC TESTS WITH PRESSURE INSTRUMENTED BLADE**
- **HOVERING ROTOR AERODYNAMIC/ACOUSTIC TESTS**
- **HOVERING ROTOR HOLOGRAMS**

PAST THEORETICAL ACCOMPLISHMENTS

The theoretical prediction techniques have been pursued along with the experimental efforts. These include determining the extent of applicability of the simple linear techniques, defining requirements for application of the non-linear (or quadrupole) terms and applying transonic aerodynamic analysis and comparing results to measured test conditions for verification.

- LINEAR ACOUSTIC PROGRAM TO PREDICT FAR-FIELD RADIATED NOISE (THICKNESS AND DISTRIBUTED FORCE)
- IDENTIFIED TRANSONIC (QUADRUPOLE) TERMS AND EXPANDED THEORY TO INCLUDE NON-LINEAR EFFECTS
- APPLICATION OF LINEAR APPROACH TO
  - UH-1H HELICOPTER (FULL SCALE) YO-3A AND OV-1C
  - AH-1 HELICOPTER (FULL SCALE) YO-3A
  - AH-64 HELICOPTER (FULL SCALE) YO-3A
  - UH-1H (MODEL IN ANECHOIC HOVER CHAMBER)
  - UH-1H (MODEL IN WIND TUNNEL)

More recent theoretical codes have permitted applications of both the steady and unsteady small disturbance transonic codes, inclusion of rotor wake geometry, and the inclusion of two dimensional vortices to more accurately define the requirements for inclusion of unsteady transonic flow field disturbances into the evaluation of the quadrupole terms. The Aeromechanics Laboratory acoustics research has also been shifting more towards the second most important noise source, rotor blade/vortex interaction noise.

- APPLICATION OF NON-LINEAR APPROACH TO UH-1H (MODEL IN ANECHOIC HOVER CHAMBER)
- NON-LIFTING TRANSONIC SMALL DISTURBANCE AERODYNAMIC SOLUTION IN HOVER
- NON-LIFTING TRANSONIC SMALL DISTURBANCE AERODYNAMIC SOLUTION IN FORWARD FLIGHT
- BLADE-VORTEX INTERACTION NOISE USING MEASURED BLADE PRESSURES

EXPERIMENTAL PROGRAMS

Current model rotor experiments are directed toward obtaining high quality simultaneous blade pressures, radiated noise and the directivity pattern. The experiments in the CEPRA-19 wind tunnel, Seclay, France, have been completed and experiments in the DNW Facility in the Netherlands are in progress. These very carefully controlled experiments are designed to provide data from which the physical process can be understood and adequately modeled. Other experimental work involves a sonic cylinder experiment designed to provide a two-dimensional verification of the non-linear high speed impulsive noise and delocalization process. Laser holography/tomography experiments are also being conducted to provide visual and quantitative data for further verification of aerodynamic noise generation and propagation mechanisms.

- **MEASUREMENT OF MODEL SCALE BLADE-VORTEX INTERACTION NOISE**
  - **CEPRA-19**
  - **DNW**
- **SONIC CYLINDER**
- **LASER HOLOGRAPHIC TOMOGRAPHY OF HIGH SPEED ROTOR**

THEORETICAL PROGRAMS

The theoretical development is currently directed toward documentation and refinement of both linear and non-linear acoustic codes. Developing simple models for blade-vortex interaction aerodynamics and noise and coupling of codes to develop adequate rotor noise prediction capabilities for the two primary sources of rotor aerodynamically generated thickness and blade-vortex interaction noise.

- **DEVELOPMENT AND DOCUMENTATION OF A FAST, SIMPLE ALGORITHM FOR LINEAR THEORY**
- **DOCUMENTATION OF QUADRUPOLE ALGORITHM FOR NON-LINEAR THEORY**
- **DEVELOPMENT OF "TRACE MACH NUMBER" (ACOUSTIC AND AERODYNAMIC) PROFILES FOR BLADE-VORTEX INTERACTION NOISE**
- **COUPLING OF 3-D TRANSONIC CODES TO HIGH SPEED ROTOR ACOUSTIC THEORY**
- **TRANSONIC CALCULATIONS OF THE SONIC CYLINDER**

TEST OBJECTIVES CEPR 19

Preliminary results of the on line AH-1G (OLS) model rotor tests in France have been partially reviewed and they contain a wealth of information on blade-vortex interaction noise and its directivity patterns. The model rotor does provide accurate scaled acoustic data and this effort has been highly successful.

- **SCALING OF BLADE-VORTEX INTERACTION NOISE**
- **MEASUREMENT OF MODEL SCALE BLADE PRESSURES AND SCALING OF LOCAL AERODYNAMICS**
- **SIMULTANEOUS BLADE PRESSURE AND NOISE MEASUREMENTS UNDER CONTROLLED AERODYNAMIC AND ANECHOIC CONDITIONS**
- **DIRECTIVITY PROFILES OF RADIATED NOISE**
- **TAPERED TIP AND AIRFOIL PROFILE CHANGES**
- **BLADE-VORTEX INTERACTION NOISE ON A 4 BLADED ROTOR**

DNW TEST OBJECTIVES

The same AH-1G (OLS) model rotor is now being tested in the DNW Wind Tunnel in the Netherlands. Test conditions in the French tunnel will be repeated in the larger tunnel with a lower turbulence level. The DNW has the capability for higher forward speed testing and the model rotor will be operated to free stream velocities of about 130 knots to provide comparisons with the full range of full scale in-flight acoustic and blade pressure measurements on the AH-1G 540 OLS rotor system.

- **SCALING OF BLADE-VORTEX INTERACTION NOISE**
- **MEASUREMENT OF MODEL SCALE BLADE PRESSURES AND SCALING OF LOCAL AERODYNAMICS**
- **SIMULTANEOUS BLADE PRESSURE AND NOISE MEASUREMENTS UNDER CONTROLLED AERODYNAMIC AND ANECHOIC CONDITIONS**
- **DIRECTIVITY PATTERNS OF RADIATED NOISE**

UP TO 80 m/sec (130 knots)

- **ROTOR NOISE PROPAGATION THROUGH A SHEAR LAYER**
- **INVESTIGATION OF DECAY LAWS**

SUMMARY

Rotorcraft noise reduction started in the early 60's and has grown steadily through the years. Industry now spends over a million dollars a year on noise reduction. Whether the government enforces noise constraint rules or not, the problem is not going to go away. The cut and try methods are not going to work but the future is bright. The primary sources of rotor noise do scale, but very careful experimentation is essential. Errors of a few percent in determining Mach number can invalidate results. Non-dimensional testing is essential. Thickness noise in principle is not an unsteady process and therefore hover testing is very useful. We are getting a good data base on blade-vortex interaction noise and its directivity patterns. Results indicate that this data will be very helpful in understanding and modeling the physics of blade-vortex interaction noise. With the ability to understand the mechanisms and predict radiated noise, creative thinking will find ways to reduce and control the major sources of rotor aerodynamically generated noise.

REQUIREMENTS

Thickness Noise

Steady small disturbance transonic codes provide the necessary input to calculate the high speed impulsive noise in hover, including the delocalization, for tip Mach numbers of interest. Unsteady small disturbance codes can be expected to provide the same capability for forward flight. For this noise source and higher frequency blade loads these codes will have to be coupled with wake codes and boundary layer codes to determine accurate load distribution and loading noise. Full potential unsteady transonic codes will only be required for the very detailed blade loading, particularly on areas of the rotor disk where the angles of attack are large enough to invalidate the small disturbance assumptions.

Blade-Vortex Interaction

Unsteady small disturbance transonic codes should be well suited to this problem. Wake prediction is essential to define the tip vortex trajectory, but high accuracy is probably not essential. The vortex will intersect the advancing blade at some rate of descent. The intensity of the noise produced should be predictable even though the accurate rate of descent when it is a maximum may not be so easily specified.

The trace Mach number is very important as it defines the Mach number of the disturbance movement in space that influences both the signal strength and direction. Specialized testing will be helpful in verifying theoretical models. These tests include wind tunnel tests of 2-D and 3-D wings with upstream vortex generators.

FACILITIES AND TOOLS

Outside hover testing for rotor high speed thickness noise can be very useful, but exceptional care is required to assure known and/or very low wind conditions. The Aeromechanics Laboratory AHC facility is a relatively low cost approach to providing less constrained hover noise measurement capability.

High speed thickness noise hover data should be very useful as a first cut at the high speed forward flight requirement, as the primary parameter is tip Mach number. The most severe problem faced by the US industry and noise research community in general is the lack of a really good rotorcraft noise test facility for speeds up to 200 kts.

The ability to scale both the high speed and blade-vortex interaction noise minimizes the facility cost and lessens the problems associated with accurate measurement requirements by LV, hot wire and the very difficult holography/tomography techniques. The constraint imposed by frequency response of microphones limits the scale to about 1/7 scale. Smaller rotors operated at full scale tip Mach numbers produce time pressure changes above the frequency response capability of conventional microphones and recording equipment.

It is essential in all model rotor experimental acoustic testing to be very careful of test techniques, and non-dimensional testing is required for scaling.

REFERENCE

1. Kittleson, John K; and Yu, Yung H.: Holographic Interferometry Techniques for Rotary Wing Aerodynamics and Noise. Presented at 1982 Army Science Conference, West Point, N.Y., June 15-18, 1982.

N84 29669

NASA/ARMY SUPPORTED NOISE SOURCE/NOISE REDUCTION PROGRAMS AT LANGLEY

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TIP-AIR-MASS INJECTION (TAMI) PROGRAM

A rotor blade interaction with a previous blade tip vortex generates a distinctive impulsive noise signature which has been labeled 'blade slap'. A series of contractual investigations focused at the blade-vortex interaction (BVI) problem were conducted by the RASA Division of the Systems Research Laboratories, Inc. (refs. 1 to 4) to test a possible technique for reducing this noise source. This technique, developed during the early 1970's, was designed to inject a high-velocity jet of air into the vortex core (see bottom of fig. 1) to cause early decay of the vortex strong velocity profile. This technique was evaluated at full scale on a hover tower and deemed promising based on smoke visualization studies of the tip vortex structure. A follow-on program was conducted at model scale in the University of Maryland Tunnel to evaluate the technique's acoustic performance at simulated forward flight (upper right of fig. 1).

The results of this program indicated that the impulsive character of the noise generated in descending low-speed flight could be greatly reduced by the use of TAMI (upper right of fig. 1). In general, it was concluded that the noise output due to blade-vortex interaction could be reduced by 4 dBA, however, with an equivalent power expenditure of approximately 14 percent of installed power. Even with a promised increase in noise reduction, this technique was not actively followed into flight test, probably due to power requirements imposed on the helicopter.

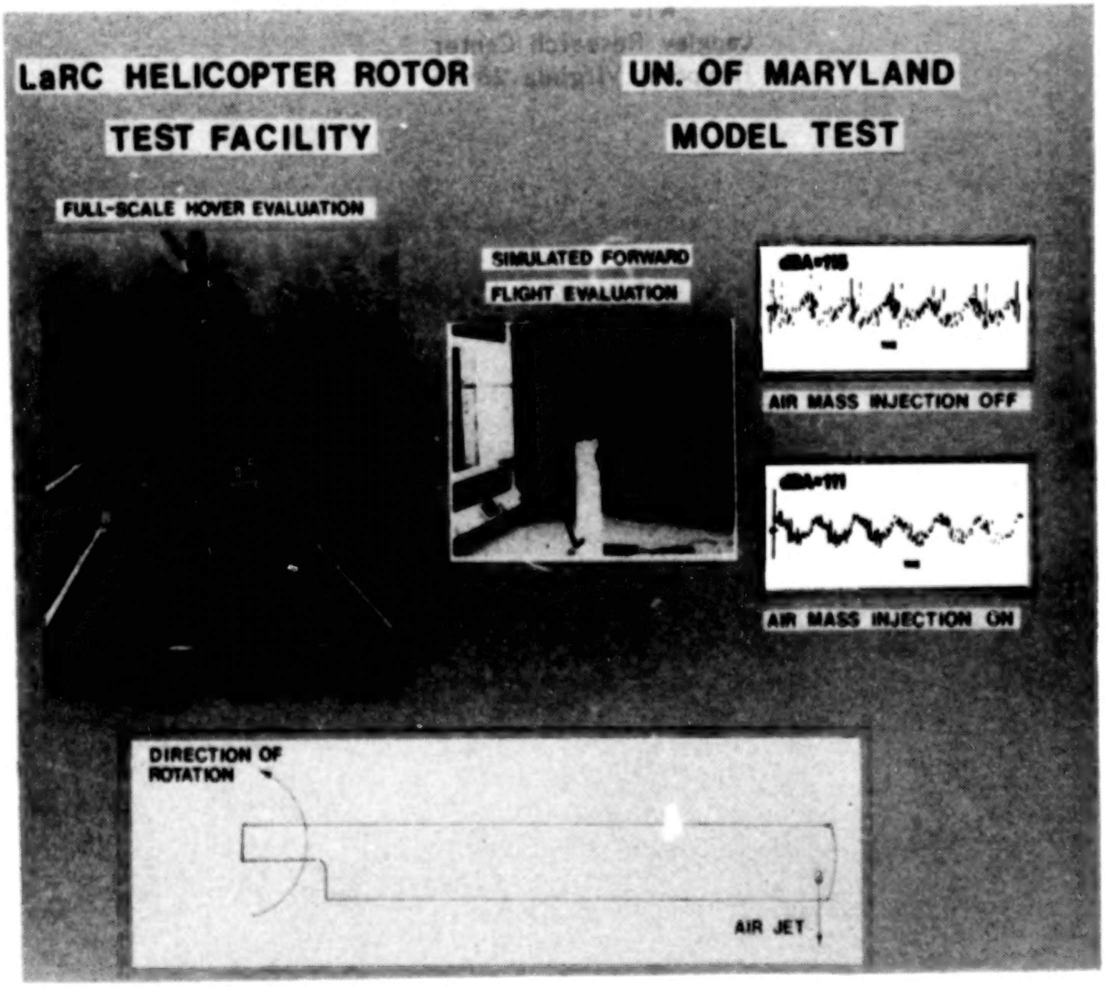
LaRC HELICOPTER ROTOR
TEST FACILITY
FULL-SCALE HOVER EVALUATION
UN. OF MARYLAND
MODEL TEST
SIMULATED FORWARD
FLIGHT EVALUATION
dBA=115
AIR MASS INJECTION OFF
dBA=111
AIR MASS INJECTION ON
DIRECTION OF
ROTATION
AIR JET

Figure 1.

MAIN ROTOR/TAIL ROTOR INTERACTION PROGRAM

A major source of helicopter noise is that produced by the tail rotor and interaction of the tail rotor with the main rotor wake. A parametric model program was initiated in the mid-1970's to define some parameters critical to this noise source. A model was constructed (ref. 5) for research programs in the Langley Acoustics and Noise Reduction Laboratory (ANRL) with adjustable tail boom height and length (see fig. 2). Variation of tail rotor position, tail rotor speed, and thrust direction in effect provided a parametric evaluation of tail rotor noise effects at various model operating conditions. The model was constructed with a 3-foot diameter rotor and installed in the jet produced by the 4-foot diameter nozzle at the ARNL, thus introducing some uncertainty in overall aerodynamic performance measurement due to boundary corrections in the relatively small tunnel.

This particular investigation (ref. 6) was conducted to determine the feasibility of using small-scale model data in identifying some of the pertinent parameters relevant to the main rotor/tail rotor interaction phenomena. The results obtained for a hover condition (lower left in fig. 2) indicated that there was a definite increase in the higher order main rotor harmonic noise due to the interaction of the tail rotor. The results obtained in forward flight conditions (lower right in fig. 2) definitely indicated an effect on noise generation due to relative position of the tail rotor.

This program was effective in identifying general parameter effects; however, due to model limitations pertaining primarily to its small size this program was not continued at the ANRL.

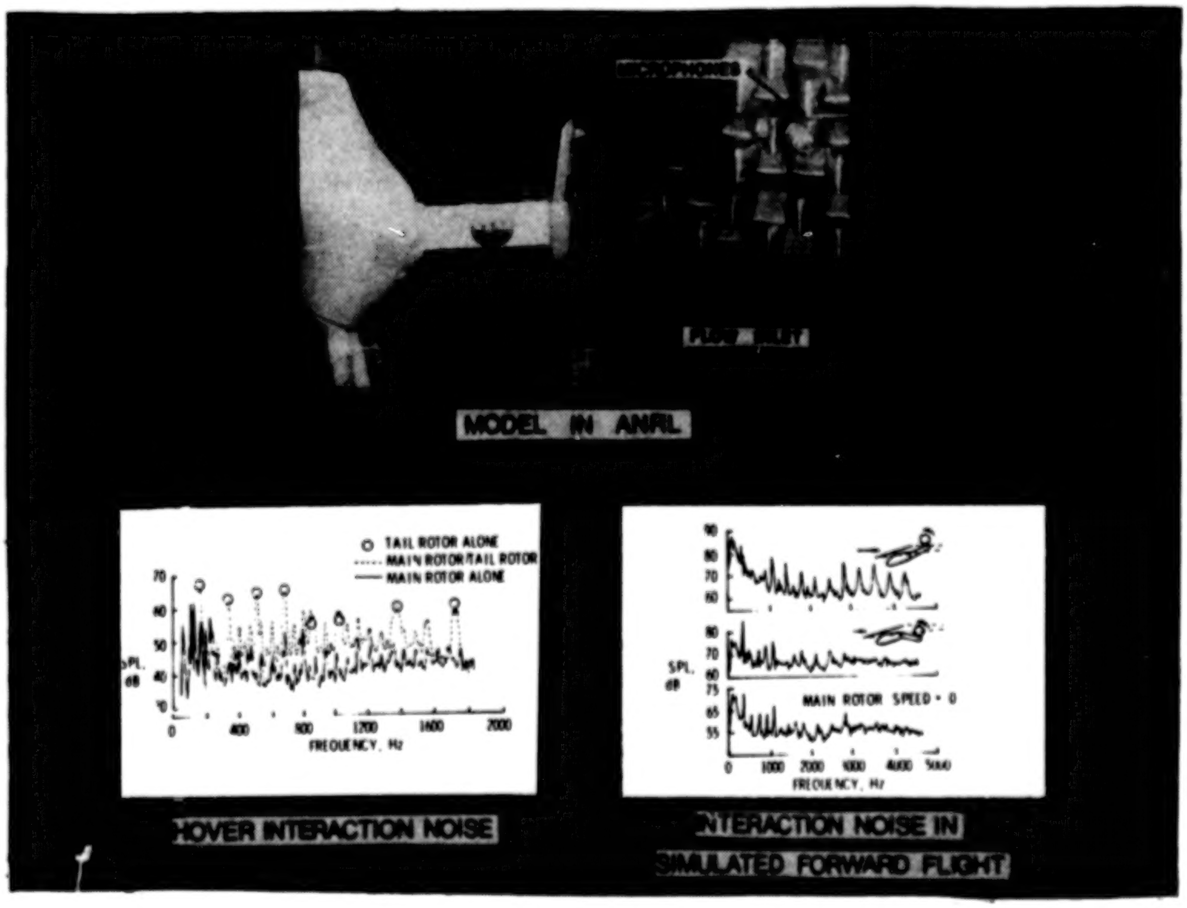

Figure 2.

M.I.T. RESEARCH ON ROTOR BROADBAND NOISE

The rotorcraft noise research at the Massachusetts Institute of Technology (M.I.T.) has been effective during the last decade in identifying noise source mechanisms and propagation characteristics. A current program supported by the Acoustics and Noise Reduction Division (ANRD) at Langley is concerned with low frequency broadband noise production due to in-flow turbulence through the rotor disk. The experimental portion of this program used a 4-foot diameter rotor system (fig. 3) in the M.I.T. 5- by 7.5-foot wind-tunnel facility (ref. 7). One aspect of this program was to study the effect of rotor tip shape on broadband noise generation (upper right of fig. 3). The test program was specifically designed to maintain certain parameters constant while varying others. In-flow turbulence changes were obtained by installation of various biplanar grids in the inlet section of the tunnel.

Some typical results from this program are presented in figure 3 showing the effect of trailing-edge sweep on the broadband noise generation from this rotor with no induced in-flow turbulence. Clearly shown in this figure are the effects of modifying the outboard tip region. Increased trailing-edge sweep reduced that portion of the noise spectrum referred to as low frequency broadband noise.

This research area is an on-going activity at M.I.T. and includes other experimental objectives. Theoretically, a noise prediction methodology was developed similar to formulations by Amiet at United Technologies Research Center. Incorporated in this method are empirical scaling expressions of peak sound pressure level with blade loading and such factors as blade flapping and twist.

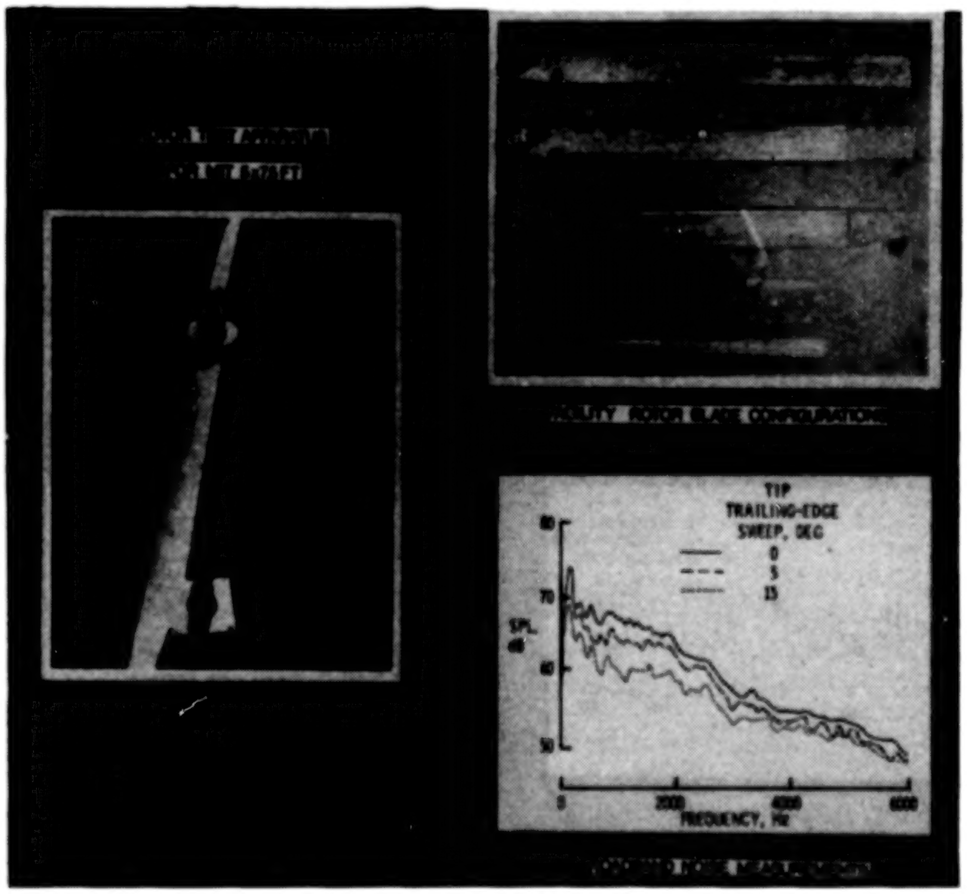

Figure 3.

UTRC RESEARCH ON TURBULENCE INGESTION NOISE

Rotorcraft noise research at the United Technologies Research Center (UTRC) has been supported by Langley for many years. Research programs by UTRC personnel have also concentrated on noise due to turbulence ingestion and identification of critical source mechanisms in this noise generation process (refs. 8 and 9). Figure 4 presents a photograph of the facility used for this research, specifically described in reference 8. Some of the results presented in this reference indicated that changes in rotor pitch were relatively ineffective in changing the generated noise characteristics. High frequency broadband noise due to turbulence ingestion was observed to be directly proportional to the number of blades on the rotor system. Furthermore, UTRC-developed prediction methodology was effective in estimating these results. Results from measurements obtained in hover indicated the requirement to measure in-flow turbulence statistics and far-field noise simultaneously to effectively predict far-field noise characteristics. Recent experimental and theoretical studies (ref. 9) demonstrated the significance of rotor blade trailing-edge noise to the total broadband noise spectrum at high frequencies. This noise mechanism is expected to control the minimum noise level generated by rotors.

A current on-going research effort at UTRC involves an experimental program designed to investigate the noise generated by the interaction of an isolated vortex with a helicopter rotor. The isolated vortex is generated by a model airfoil mounted upstream of the rotor. In this program, detailed flow-field measurements in the vortex field will help to define noise source mechanism effects and improve the state of the art in prediction of far-field noise generation without reliance on empirically derived functions.

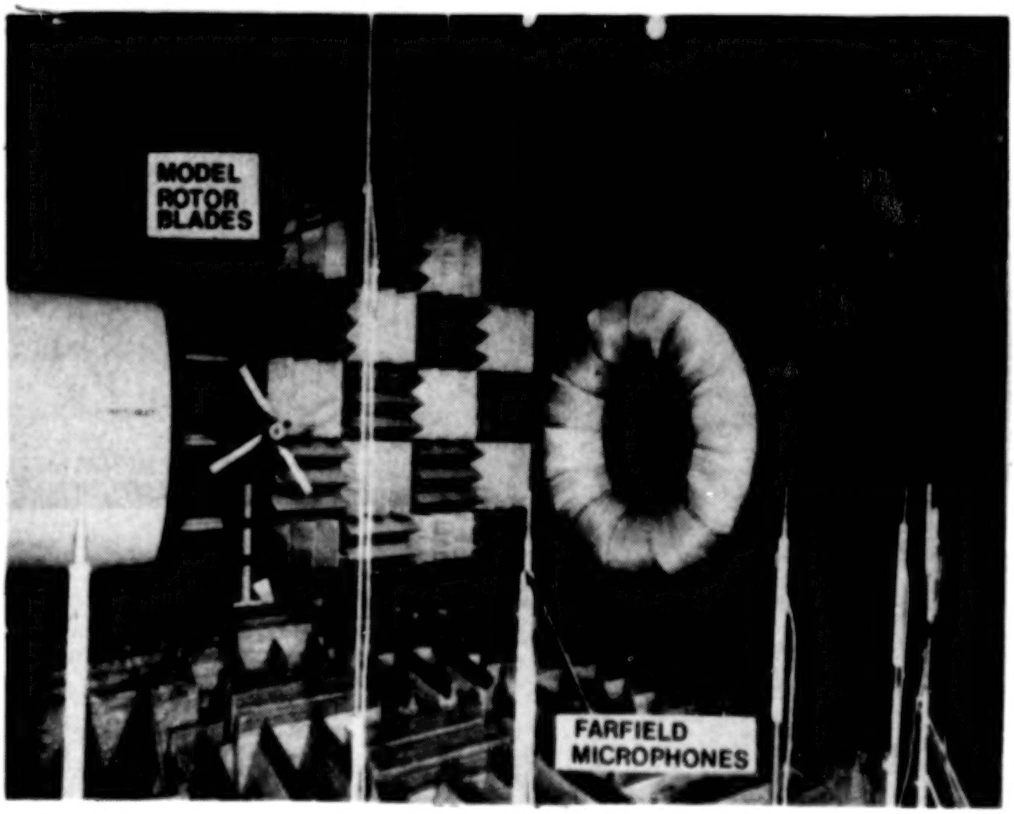

Figure 4.

ANRL ROTOR BLADE SELF NOISE

A current in-house program has been conducted to evaluate the noise source mechanisms related to generation of noise from an isolated stationary airfoil in the Acoustics and Noise Reduction laboratory (ANRL) at Langley (upper left portion of fig. 5). The objectives of this program are to identify the particuar noise generation process of attached or separated boundary layers and three-dimensional turbulence effects in the tip vortex formation region. Particular interest is focused on aerodynamic and acoustic scaling effects of velocity and different airfoil chord lengths (lower left portion of fig. 5), which range in this test from 1 inch to 24 inches.

Near wake-turbulence measurements have been obtained in detail by a hot wire mounted to a computer controlled traversing rig (upper right of fig. 5). These turbulence characteristics were associated with far-field noise measurements in standard coherence analysis procedures.

A recent program conducted at the ANRL demonstrated the noise generation mechanism attributable to trailing-edge noise effects (ref. 10). Some results of this program are presented in the lower right portion of figure 5. This investigation, according to the results in figure 5 and reference 10, demonstrated the effect of trailing-edge thickness on the acoustic spectrum obtained from acoustic measurements. The thicker trailing edge, which was obtained by modification of the airfoil used for the thin trailing edge, generated a higher noise level than that for the thin trailing edge in the higher frequency region of the overall spectrum. These results (obtained from the present research program and through cooperation with UTRC) have provided valuable insights to the importance of rotor blade self noise in the overall spectrum of rotor noise.

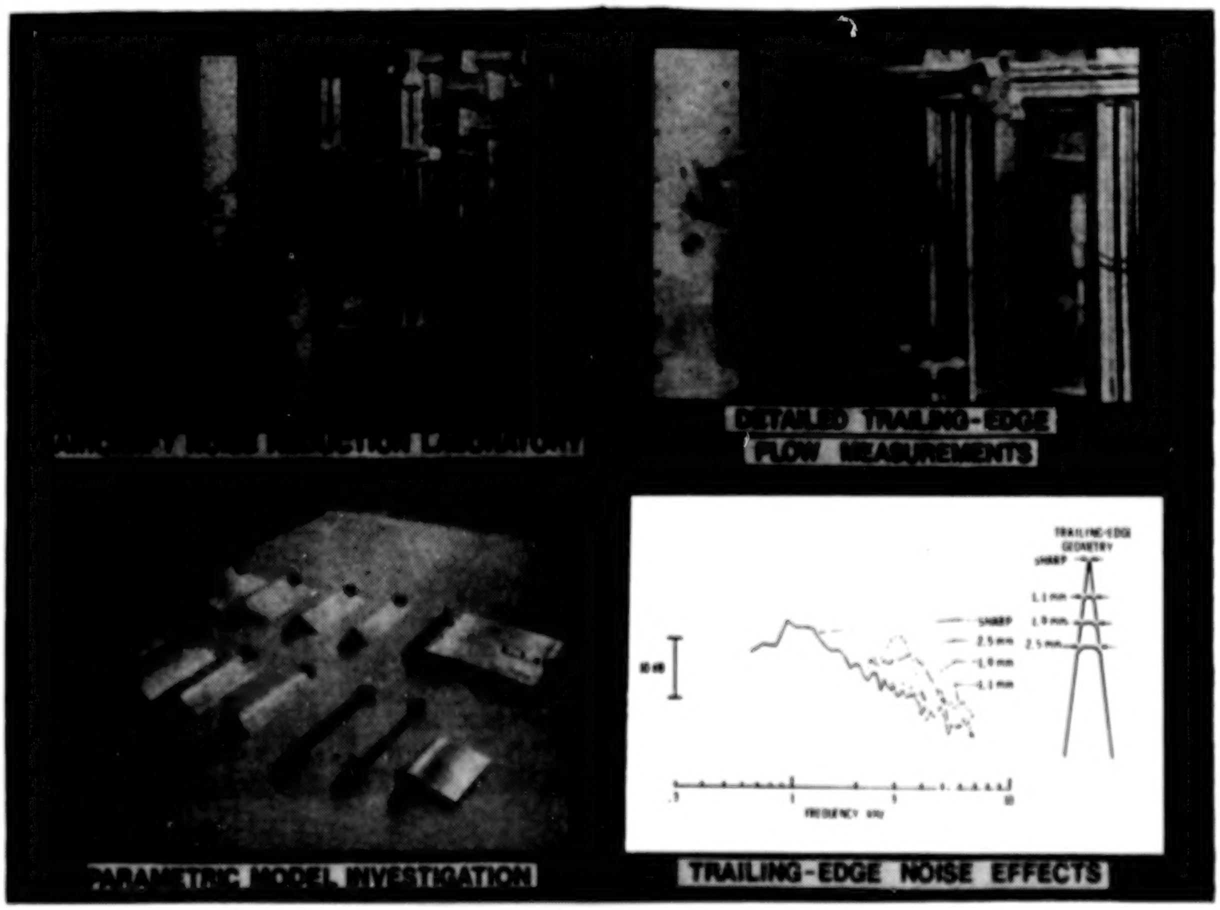
DETAILED TRAILING-EDGE FLOW MEASUREMENTS
PARAMETRIC MODEL INVESTIGATION
TRAILING-EDGE NOISE EFFECTS
TRAILING-EDGE GEOMETRY
SHARP
1.1 mm
1.9 mm
2.5 mm
SHARP
2.5 mm
1.9 mm
1.1 mm
10 dB
.3
1
3
10
FREQUENCY kHz

Figure 5.

FULL-SCALE OGEE-TIP ROTOR ACOUSTIC TESTS

A flight test evaluation was conducted in the mid-1970's to evaluate the effect of a rotor blade tip designed to diffuse the rotor blade tip vortex. This ogee-tip modification (lower left of fig. 6) was considered promising as a noise reduction scheme directed primarily at blade vortex interaction (BVI) noise. Based on encouraging results of vortex flow studies, pressure data, and aerodynamic performance evaluations, a modified set of rotor blades was designed and constructed for flight tests on the UH-1H helicopter (upper portion of fig. 6).

Aerodynamic performance measurements obtained in this program as reported in reference 11 indicated a significant increase in forward-flight performance and hover. Oscillatory control loads were reduced by as much as 50 percent by the ogee tip. Rotational noise in hover and far-field noise in forward flight were reduced by this tip modification. The far-field noise reduction was attributed to decrease in the compressibility impulsive noise due to the thin profile of the ogee tip.

Near-field measurements obtained with a microphone mounted near the fuselage under the rotor were acquired for many flight conditions tested. This BVI noise was reduced by as much as 15 dB. A more general evaluation of the effect of this tip modification was presented in reference 12 and again here in the lower right of figure 7. Peak levels of the near-field impulsive noise below each rotor were found at conditions indicated. The BVI noise conditions for the standard rotor were moved to higher rates of descent for a given air speed through use of the ogee tip. The extent of the flight envelope covered by this intense noise characteristic was greatly reduced and the maximum intensity of the ogee tip impulsive noise below the rotor was signficantly lower than the maximum noise of the standard rotor for this flight condition.

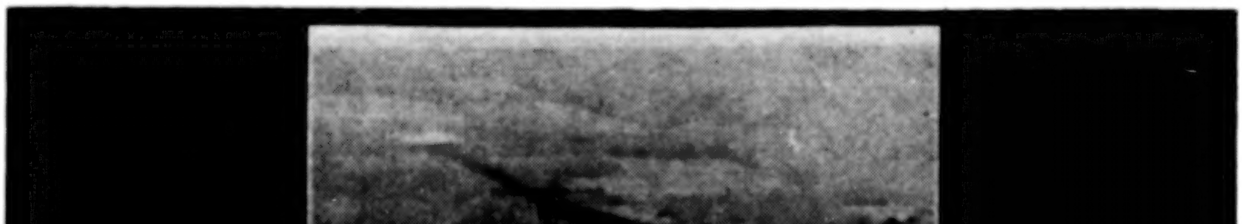
ROTOR BLADES TESTED
UH-1H STANDARD TIP BLADE
DIRECTION OF ROTATION
UH-1H OGEE TIP BLADE
PEAK LEVELS OF NEAR-FIELD IMPULSIVE NOISE
V, KNOTS
0
20
40
60
80
100
120
RATE OF DESCENT, FT/MIN
200
400
600
800
1000
STANDARD TIP
OGEE TIP

Figure 6.

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BLADE VORTEX INTERACTION NOISE REDUCTION BY ROTOR BLADE TIP MODIFICATION

Encouraging results obtained on different rotor configurations in reducing blade vortex interaction (BVI) noise by rotor blade tip modification precipitated a scale model investigation in the Langley 4- by 7-Meter Tunnel (fig. 7). This in-house program included one set of rotor blades for a four-bladed general research model system, with tip-change capability. Four tip configurations were selected for comparison with the standard square tip (upper right of fig. 7) based on results obtained in other research programs: the ogee tip (ref. 11), the subwing tip (ref. 13), the endplate tip (ref. 14), and the swept tip (ref. 15).

Aerodynamic performance comparisons of these rotor systems obtained concurrently with the acoustic measurements were reported in reference 16. The results indicated a reduction in power required was possible with the ogee tip and swept tip as compared with the square tip rotor. The subwing and endplate tip required more power than the square tip at the same thrust coefficient.

Some acoustic results presented in reference 12 are included in the lower right portion of figure 7. These results are presented as peak-to-peak level of the impulsive waveform generated as a function of simulated descent angle for a sample tunnel free-stream velocity condition. These results demonstrate the conclusions drawn in reference 12 that the subwing and ogee tips were most effective in reducing the impulsive signature of BVI noise. The swept tip was effective in this reduction but to a lesser degree than the ogee and subwing tips. Further analysis of these data (ref. 16) identified the source of the impulsive noise in the rotor disk to be between 65° and 90° azimuth and 0.6 to 1.0 radii from the rotor hub. These source location calculations were obtained using the data from three or more microphone locations for all tip modifications tested.

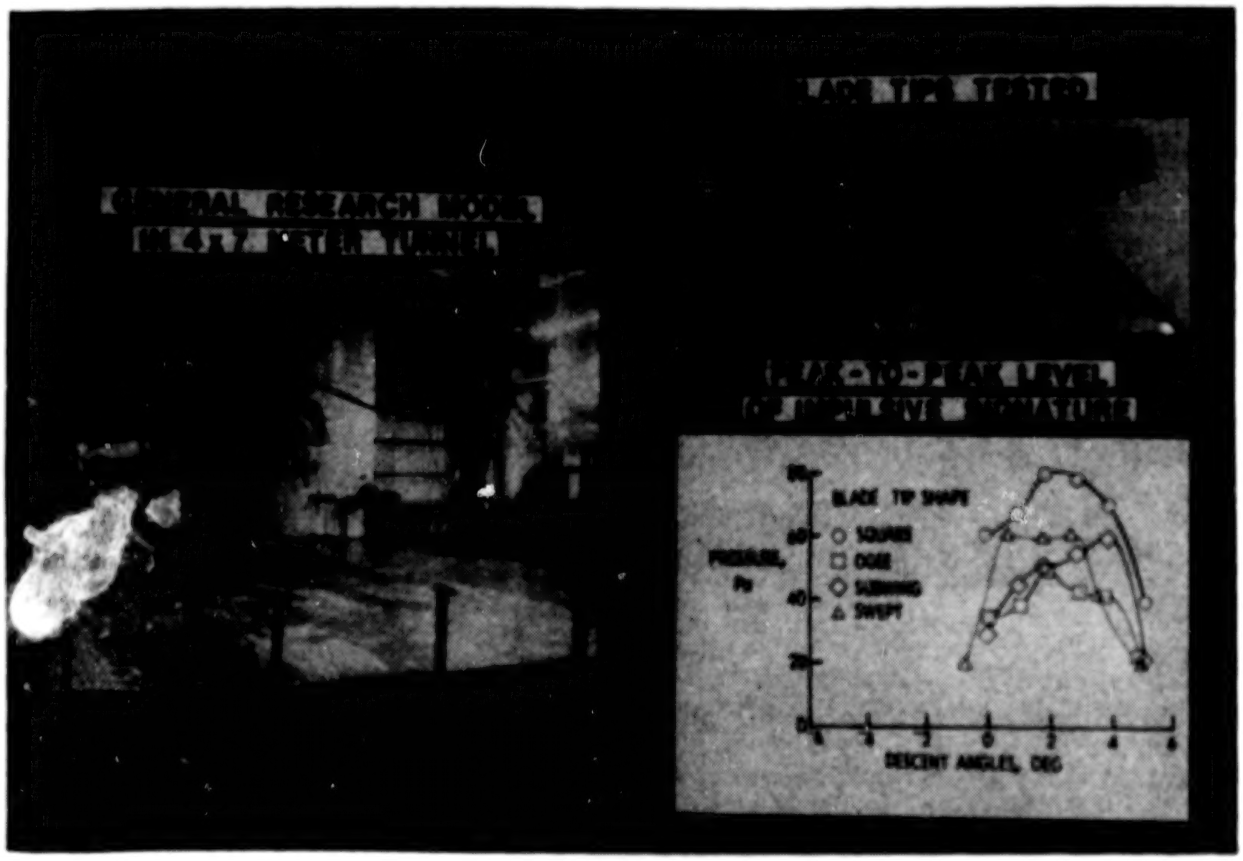

Figure 7.

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AH-1 ACOUSTIC TESTS

One of the first model helicopter noise programs conducted in the Langley 4- by 7-Meter Tunnel was performed in 1977. This program (upper left portion of fig. 8) was conducted with the specific objectives of determining if blade vortex interaction (BVI) noise could be generated at model scale in a wind-tunnel facility and how well the model results were typical of flight results. The flight test (upper right portion on fig. 8) results (ref. 17) were conducted under contract to the Langley Research Center using near-field microphones mounted near to the fuselage on a nose boom and right wing boom of an AH-1 helicopter. Microphones were installed in the model test in positions scaled to those for which the flight test data were available. Model and tunnel conditions were carefully set to properly scaled flight conditions known at the time. Sample results of this program as reported in reference 18 are presented for scale model tests in the lower left portion of figure 8 and for flight tests in the lower right portion of figure 8. Pressure time histories obtained on both nose and right wing microphones are presented in each case.

These data demonstrate the conclusions developed in reference 8 that the occurence and location of the BVI noise signature relative to the 1/rev signal from the rotor were similar to those recorded in flight. Flight test results at different altitudes, which were noted as 'smooth air' and 'bumpy air', for very similar flight conditions suggested a turbulence effect on the amplitude of the BVI noise signature. The good agreement between model and flight results at tunnel high turbulence level and 'bumpy air' conditions respectively suggested that the turbulence in-flow to the rotor system can alter the intensity and occurrence of this unique form of noise generation.

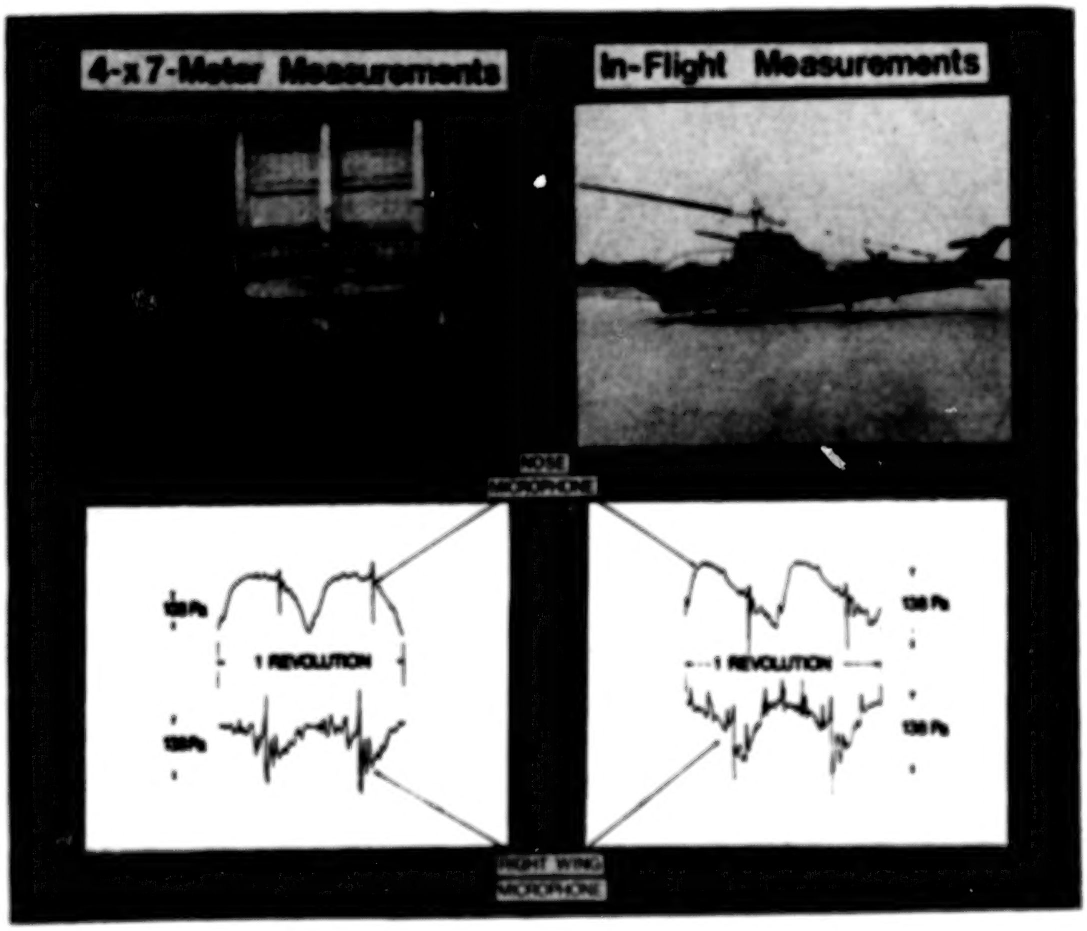

Figure 8.

IN-FLIGHT AH-1G ACOUSTIC MEASUREMENTS WITH YO-3A

The flight test program of the AH-1 rotor blades used in the tests reported in reference 17 has been repeated with improvements to the rotor blade surface pressure measurement system (fig. 9). This new program was conceived and conducted as a joint program between the Ames and Langley Research Centers. Instead of near-field fuselage mounted acoustic measurements, the unique in-flight measurement technique with the YO-3A aircraft, developed by Army Ames personnel, was employed. Performance and acoustic data were acquired, including the blade pressure measurements, the air-to-air acoustic measurements (YO-3A) and air-to-ground acoustic measurements at Crow's Landing in California.

The acoustic data reduction has been initiated by the Langley Research Center's Acoustics and Noise Reduction Division personnel. The acoustic data base in combination with the blade pressure data will be used to improve the basic state-of-the-art knowledge of source mechanisms related to helicopters and to improve and validate the Langley helicopter noise prediction capability.

Design and construction of a one-quarter scale model rotor system has begun for acoustic tests in the Langley 4- by 7-Meter Tunnel. The results of this program and similar tests by Army Ames personnel at one-seventh scale in more anechoic facilities in Europe will provide further data with which to validate prediction methodology and to establish the effects of scale on noise.

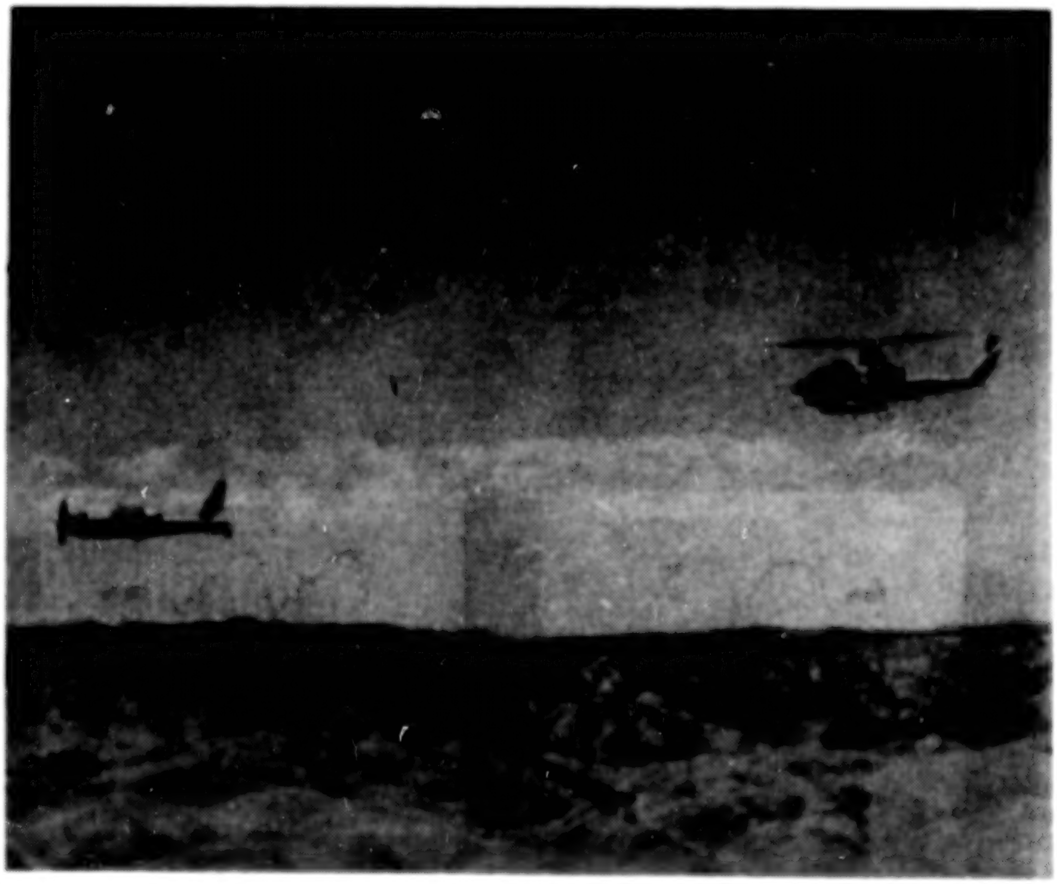

Figure 9.

UH-1H ACOUSTIC TESTS

An advanced rotor system has been designed for the UH-1H main rotor helicopters for the U.S. Army. An evaluation of the aerodynamic (ref. 19) and acoustic (ref. 20) performance of this new rotor system for this vehicle has been completed using a one-quarter scale model in the 4- by 7-Meter Tunnel (upper left portion of fig. 10). The standard UH-1H rotor blades used for this program were carefully scaled dynamically to the UH-1H flight blades. A set of advanced rotor blades were also constructed with dynamic characteristics as close as possible to the standard rotor blades. Results presented in reference 19 demonstrated a 10 percent improvement in hover performance and a 17 percent improvement in forward flight performance. Concurrently, in-plane high-speed impulsive noise measurements indicated a reduction of as much as 8 dB. The lower left portion of figure 10 indicates typical result in the time-history comparison of the acoustical comparison of the standard and advanced rotor system. The large negative waveform evident in the standard rotor system data (typical of high-speed impulsive noise) was greatly reduced (as much as 56 percent) by the advanced rotor system.

In-flight near-field measurements obtained during the investigation described in reference 11 (upper right portion of fig. 10) provided acoustic data at flight conditions carefully modeled in this model test program. The data presented in the lowest right portion of figure 10 is typical of measurements obtained specifically to examine model scale effects. The model data have been Strouhal scaled in time and corrected for density differences to account for the state-of-the-art knowledge of scale effects. These data were also expanded in amplitude to display the blade vortex interaction (BVI) wave evident at each blade passage. The low frequency loading noise correlation between flight and model data agreed very well even in the frequency domain up to about the fourth harmonic of the blade passage frequency. However, the character of the BVI waveform in this case was not modeled in the tunnel particularly in amplitude, yet the location of the apparent blade vortex encounter in the rotor disk seems to have been preserved. The early AH-1 model tests in the 4- by 7-Meter Tunnel demonstrated the importance of in-flow turbulence characteristics in the generation of this noise characteristic. Further research in this area of helicopter noise is required to determine the performance parameters responsible (model scale, microphone location, turbulence, rotor performance, etc) for this apparent anomaly.

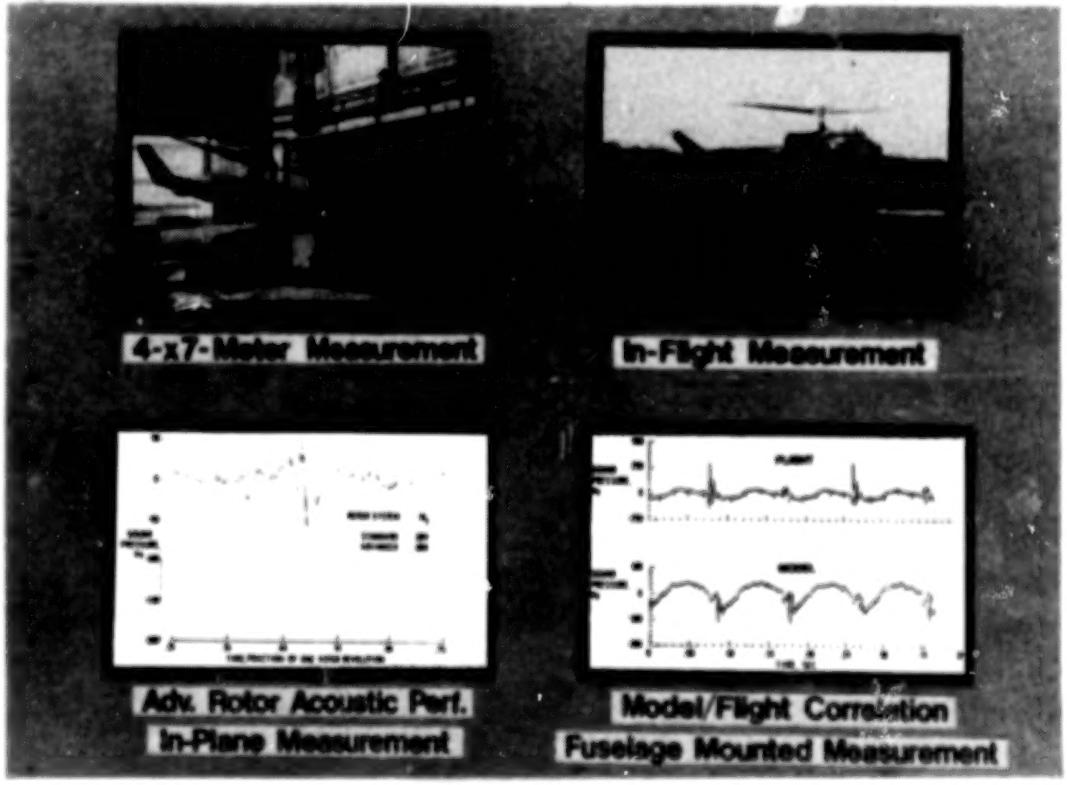

Figure 10.

SUMMARY

In summary, the helicopter noise research related to noise source mechanism identification and reduction has included many of the critical noise problems experienced by the helicopter. These include blade-vortex interaction (BVI) noise, broadband turbulence ingestion noise, rotor blade self noise including trailing-edge effects, model scale effects evaluations, and to some degree main rotor/tail rotor interaction noise. Issues that seem to arise from this brief evaluation of Langley's experience are:

1. Broadband noise can be a significant contribution to the overall noise problem and future rotor noise research should be encouraged.

2. Scale model investigations are an effective means of conducting helicopter noise research; however, more model/flight correlation studies are required to develop a high degree of confidence of the use of scale model results in the design process of helicopters.

3. Main rotor/tail rotor noise has been examined only briefly at Langley; however, more detailed investigations identifying critical factors affecting this very important noise mechanism are required.

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SESSION II

SCALING LAWS AND EMPIRICAL NOISE PREDICTION

Chairman:	H. A. Morse Aeromechanics Laboratory, AVRADCOM
Vice-chairman:	C. R. Cox Bell Helicopter Textron

N84 29670

ROLE OF EMPIRICAL METHODS

H. Sternfeld
Boeing Vertol Company
Philadelphia, Pennsylvania

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INTRODUCTION

There are different levels of helicopter noise prediction which may be appropriate at various stages in the design process. In the early preliminary design stages, when available information is usually limited to parameters such as gross weight, tip speed, forward speed, rotor radius and possibly number of blades, one is limited to purely empirically based methodology. As the design progresses, and airfoil blade planforms and twists are defined, predictions of airloads, vortex paths, and compressibility effects may permit application of more analytically based sound pressure level prediction methods. At the present stage of development of first principle prediction methodology, however, the designer may still find it necessary to supplement such analyses with modifications based on empirical experience.

The application of wind tunnel models is also developing into a useful tool for predicting full scale helicopter noise (Figure 1).

HELICOPTER NOISE PREDICTION

EMPIRICAL PREDICTIONS - PRELIMINARY DESIGN ESTIMATES

- LIMITED CONFIGURATION DEFINITION

ANALYTICALLY BASED PREDICTIONS WITH EMPIRICAL MODIFICATIONS - CONFIGURATION DEVELOPMENT -

- REQUIRES DETAILED DEFINITION OF:
  - AIRFOILS
  - PLANFORM
  - AIRLOADS
  - VORTEX INTERACTION EFFECTS .....

WIND TUNNEL MODELS - CONFIGURATION DEVELOPMENT

- ESTABLISH COMPARATIVE TRENDS
  - AIRFOILS
  - TIP SHAPES
  - ROTOR SPACING .....
- PREDICT ABSOLUTE FULL SCALE LEVELS

Figure 1

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SINGLE PARAMETRIC VARIATION

It is not uncommon to find empirical data presented as variations in level as a function of a single parameter. Care must be taken, however, to ensure that there are not other significant variables acting. Figure 2, which is in the format of helicopter noise standards, includes many different helicopters operating at different tip speeds, and even with different basic configurations. Such a chart should not be used either to predict noise levels or to establish growth trends for a particular model.

Figure 3, on the other hand, which was derived from data on a particular helicopter where the only variable was rotor tip speed, has a standard deviation of only 1 dB, which reflects test repeatability. Such data can be used both for establishing trends and to test analytical predictions.

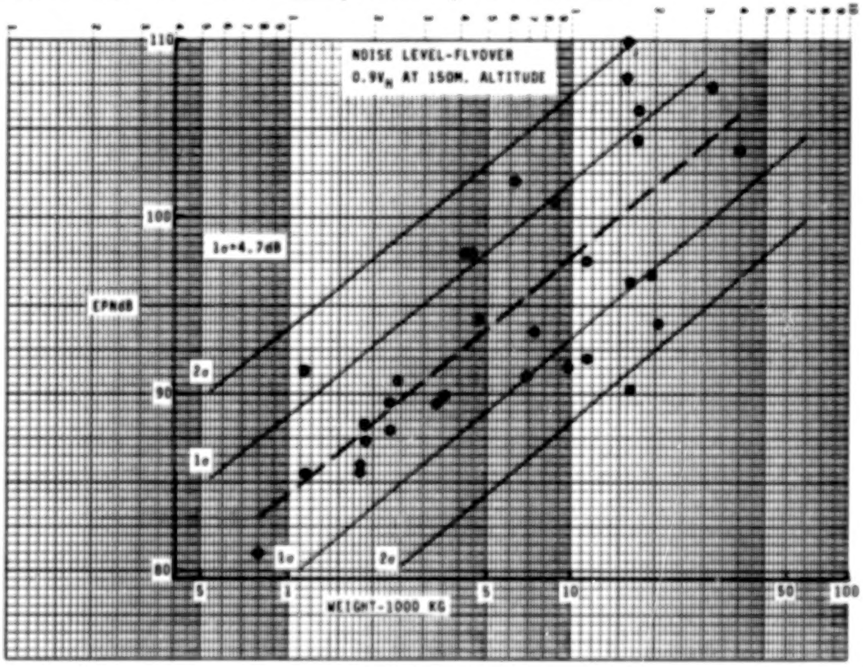

Figure 2

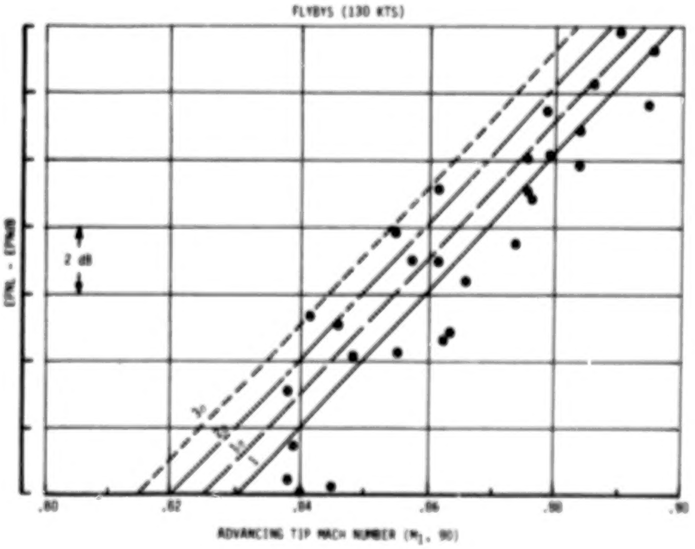

Figure 3

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NOISE PREDICTION PROGRAM

Because a helicopter noise signature is comprised of many elements and the prediction methodology for each **is** constantly changing, Boeing Vertol has developed a modular computer program (HELicopter NOise Prediction - HELNOP) (Figure 4) constructed so that the prediction methodology for any noise component can be modified or repaced without upsetting the other components. These methods may be first principle, purely empirical, or a mixture. At the present time each module employs some degree of empiricism in its input.

Once the program has predicted the acoustical spectrum at an initial location, it can move the aircraft in one-half-second increments along any flight path and provide results in several formats shown in Figure 5.

HELICOPTER EXTERNAL NOISE SOURCES INCLUDED IN
COMPREHENSIVE NOISE PREDICTION PROGRAM(R84)

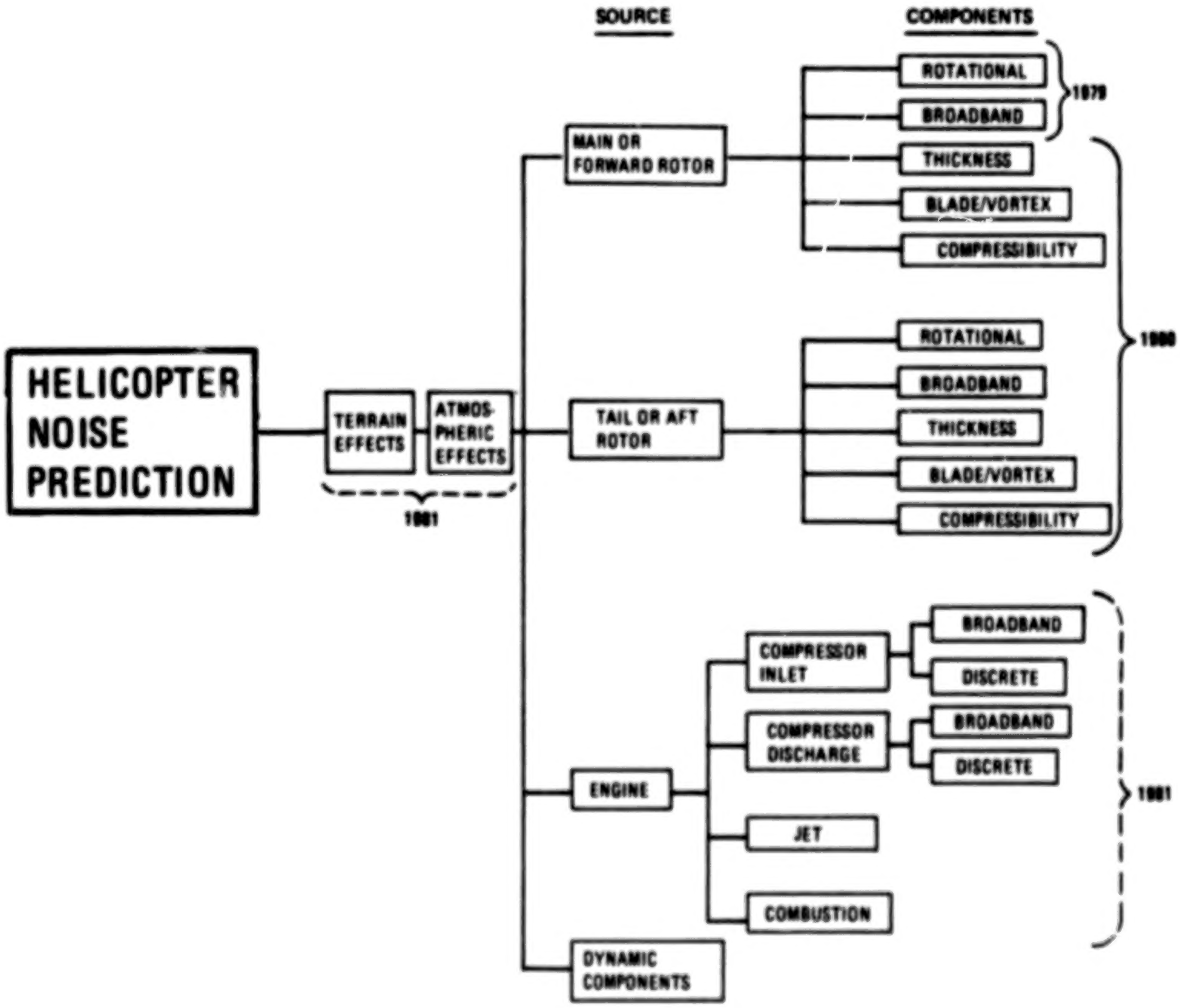
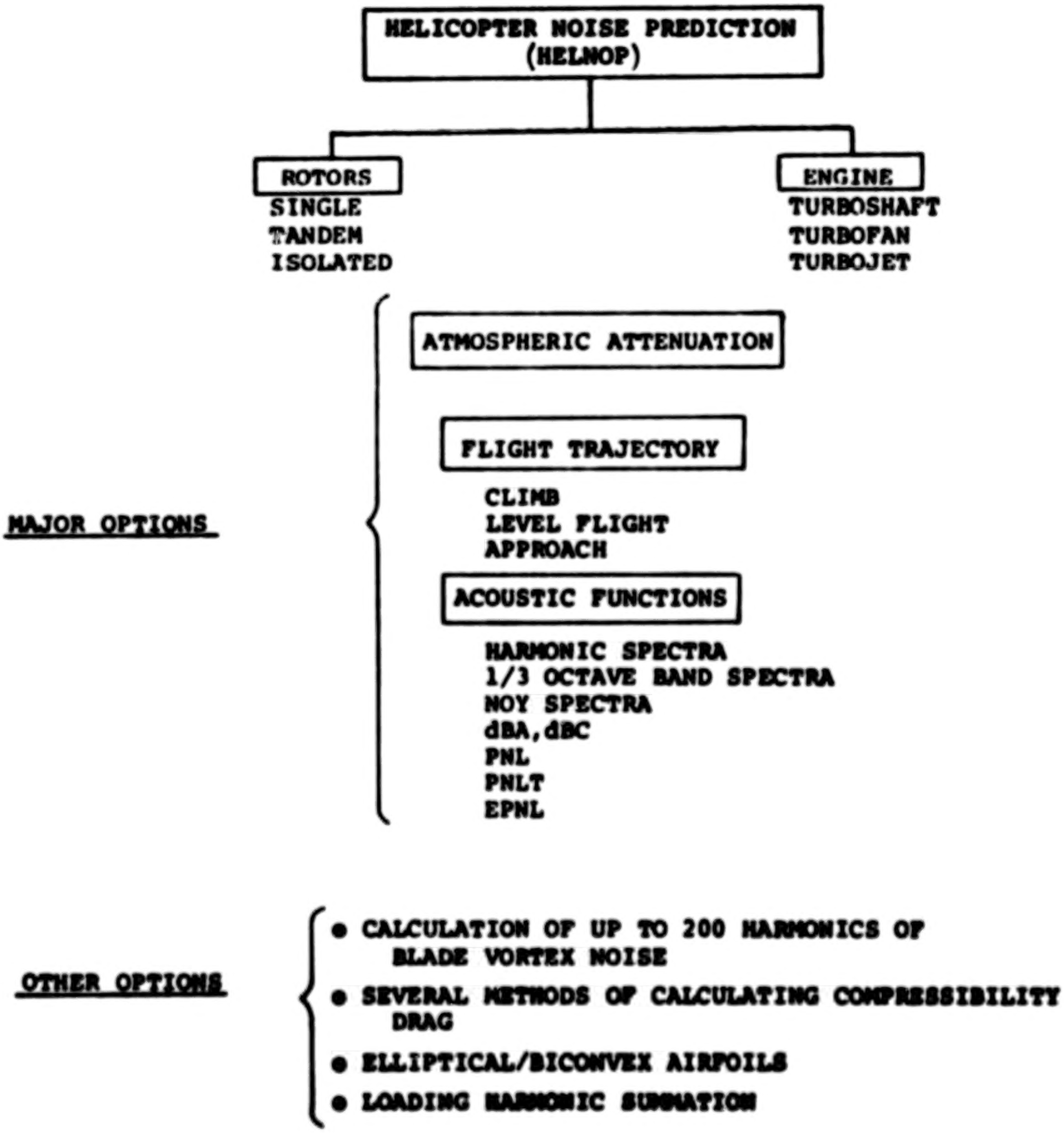

SOURCE
COMPONENTS
HELICOPTER NOISE PREDICTION
TERRAIN EFFECTS
ATMOSPHERIC EFFECTS
1981
MAIN OR FORWARD ROTOR
ROTATIONAL
BROADBAND
1979
THICKNESS
BLADE/VORTEX
COMPRESSIBILITY
TAIL OR AFT ROTOR
ROTATIONAL
BROADBAND
THICKNESS
BLADE/VORTEX
COMPRESSIBILITY
1980
ENGINE
COMPRESSOR INLET
BROADBAND
DISCRETE
COMPRESSOR DISCHARGE
BROADBAND
DISCRETE
JET
COMBUSTION
1981
DYNAMIC COMPONENTS

Figure 4



HELICOPTER NOISE PREDICTION (HELNOP)

ROTORS
- SINGLE
- TANDEM
- ISOLATED

ENGINE
- TURBOSHAFT
- TURBOFAN
- TURBOJET

MAJOR OPTIONS

ATMOSPHERIC ATTENUATION

FLIGHT TRAJECTORY
- CLIMB
- LEVEL FLIGHT
- APPROACH

ACOUSTIC FUNCTIONS
- HARMONIC SPECTRA
- 1/3 OCTAVE BAND SPECTRA
- NOY SPECTRA
- dBA, dBC
- PNL
- PNLT
- EPNL

OTHER OPTIONS
- CALCULATION OF UP TO 200 HARMONICS OF BLADE VORTEX NOISE
- SEVERAL METHODS OF CALCULATING COMPRESSIBILITY DRAG
- ELLIPTICAL/BICONVEX AIRFOILS
- LOADING HARMONIC SUMMATION

Figure 5

ROTATIONAL NOISE

Rotational noise is predicted by the method of Lowson and Ollerhead (ref. 1). (See Figure 7.) Although essentially a first principle method, the prediction of noise over an adequate range requires definition of high harmonic airloads beyond the capability of any fluid dynamicist. Predicted first harmonic airloads are used and the amplitudes of succeeding harmonic airloads are derived from an assumed decay rate which may be based on the relatively small amount of available measured pressure data. Figure 7 illustrates the empirical derivation of an airload harmonic decay rate of 1.3 from measured blade pressure data while Figure 8 shows the improvement in prediction achieved by applying the decay exponent of 2.0 recommended in reference 1.

$$pmB = \sum_{\lambda=o}^{\infty} K \cdot \frac{T}{Rr\ \lambda^{k}} \quad (C_{\lambda_T}\ nM_e \sin\theta)J_1' - C_{\lambda_D}\ J_2' - (C_{\lambda_C}\ nM_e \cos\theta)J_3'$$

$$K = \frac{1}{4\pi\sqrt{2}}$$

T = rotor thrust

R = radius of action of blade forces

r = distance from rotor center to field point

λ = air loading harmonic number

k = loading power low exponent

C_{λ_T}, C_{λ_D}, C_{λ_C} = thrust, drag, radial force harmonic coefficients

n = mB, sound harmonic number

M = rotational Mach number

θ = angle between disc plane and observer

J_i' = complex collection of Bessel functions of argument ($nM \cos\theta$)

Figure 6

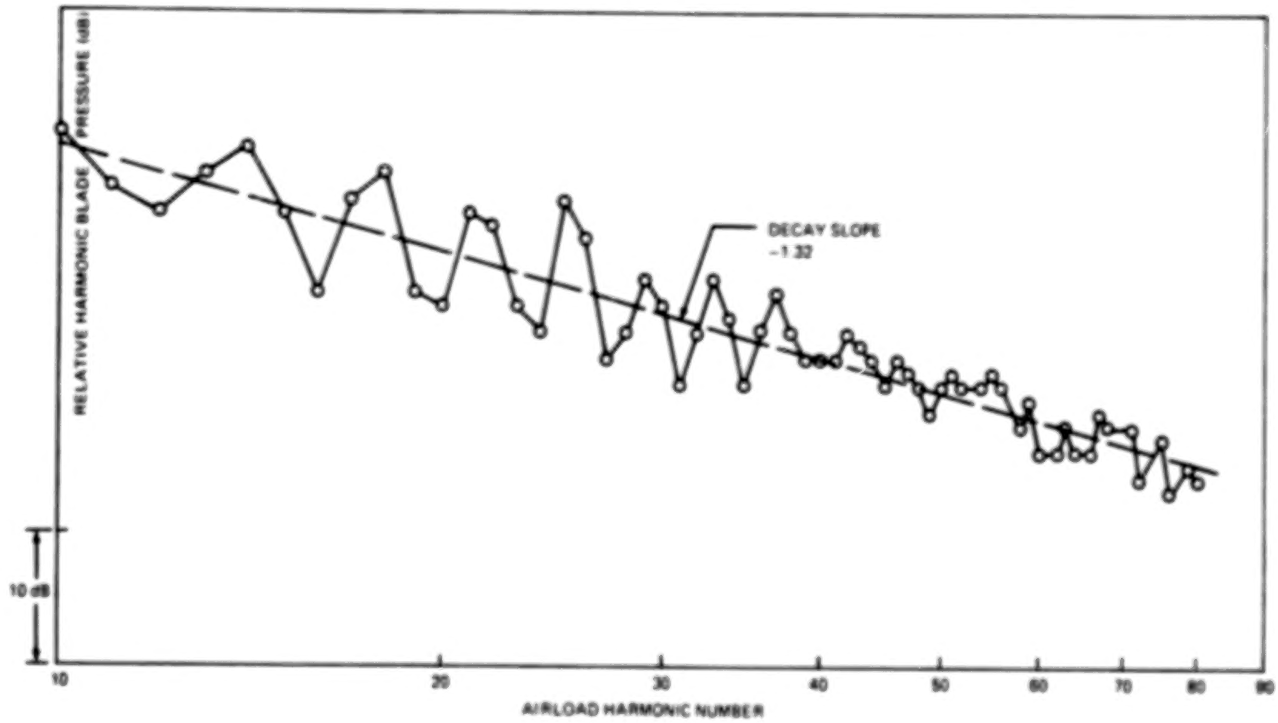


Figure 7

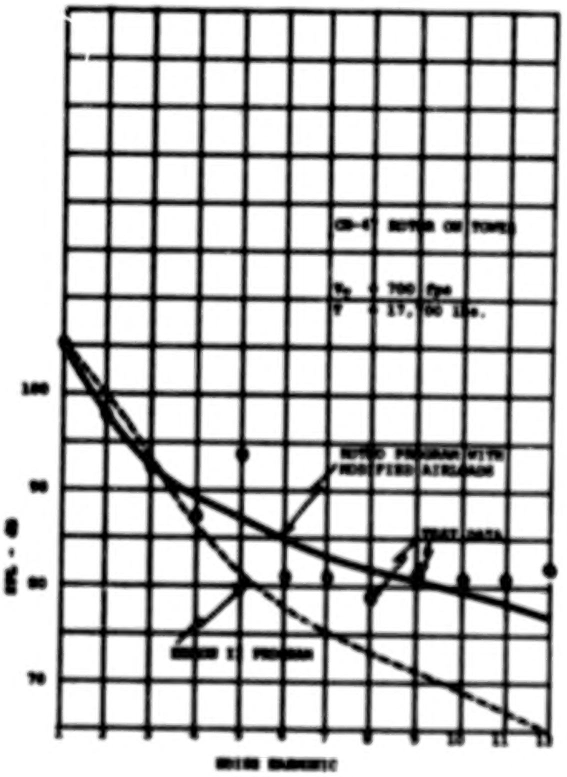


Figure 8

BLADE-VORTEX INTERACTION NOISE

The intersection of a rotor blade with a vortex filament trailed by a previous blade results in a fluctuating airload on the following blade. The amplitude of the disturbance is related to the intersection angle between the vortex and blade, the relative velocities at the blade section at intersection, and the dimensions and strength of the vortex. Most of these variables are not well known for any flight condition and assumptions regarding the input values must be made. For example, the analytical expression shown in Figure 9 (ref. 2) models the vortex as a theoretical gust profile in the formulation of the harmonic content of the pressures. In addition, lacking an accurate high harmonic airload prediction capability (particularly for vortex interactions) assumptions must be made regarding fluctuating airloads and effective dimension of the vortex at intersection. Figure 10 illustrates the empirical adjustment required to the analytical expression to obtain reasonable agreement with measurements. Without these adjustments, noise prediction is significantly less accurate.

$$p_{mB} = K_T \left(\frac{\Delta L}{L_o} E \rho_w \right) mB \chi_s$$

$$K_T = \frac{T M_e}{2\pi r R} \sin \sigma \qquad \chi_s = \frac{\sin (mBE\rho_w - 1)}{4(mBE\rho_w - 1)} - \frac{\sin (mBE\rho_w + 1}{4(mBE\rho_w + 1)}$$

$\frac{\Delta L}{L_o}$ = fractional steady load change per blade

E = number of interactions per revolution

ρ_w = load solidity (fraction of effective disk annulus occupied by the unsteady loading region, $\Delta\psi/360$

K_T = directivity term

T = rotor thrust

M_e = $\frac{M_t}{1-M_f \cos \theta}$

R = blade radius

r = distance from source to observer

χ_s = blade loading spectrum function

Figure 9

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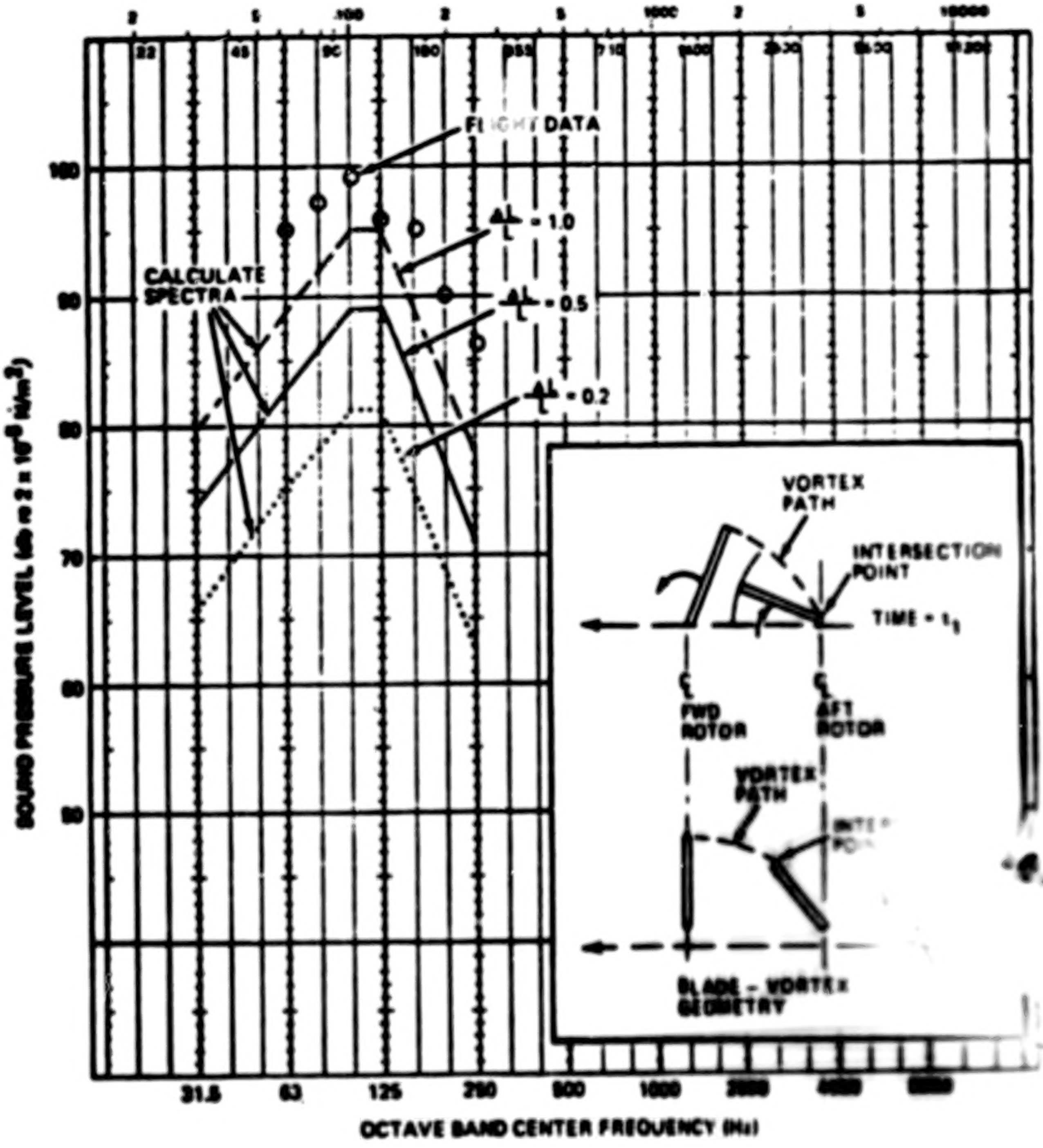

Figure 10

THICKNESS NOISE

Rotors operating at high advancing tip speeds display a rapid buildup in impulsive noise. Several analytical procedures for calculating thickness noise exist which do not require empirically derived input for their solution. To date however prediction of the generated sound pressure levels has not been sufficiently accurate in the higher speed range (Figure 11) of concern to modern helicopters. An empirically based adjustment, which was derived from flight test data (ref. 3) and is shown in Figure 12, is currently used at Boeing Vertol in preference to any analytical procedure (ref. 4). Note the strong similarity in shape to the data of Figure 11.

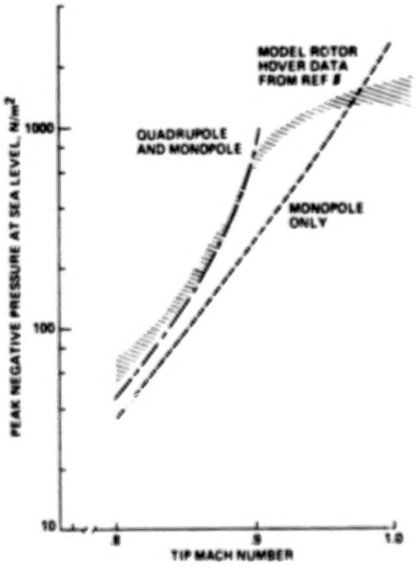

Figure 11

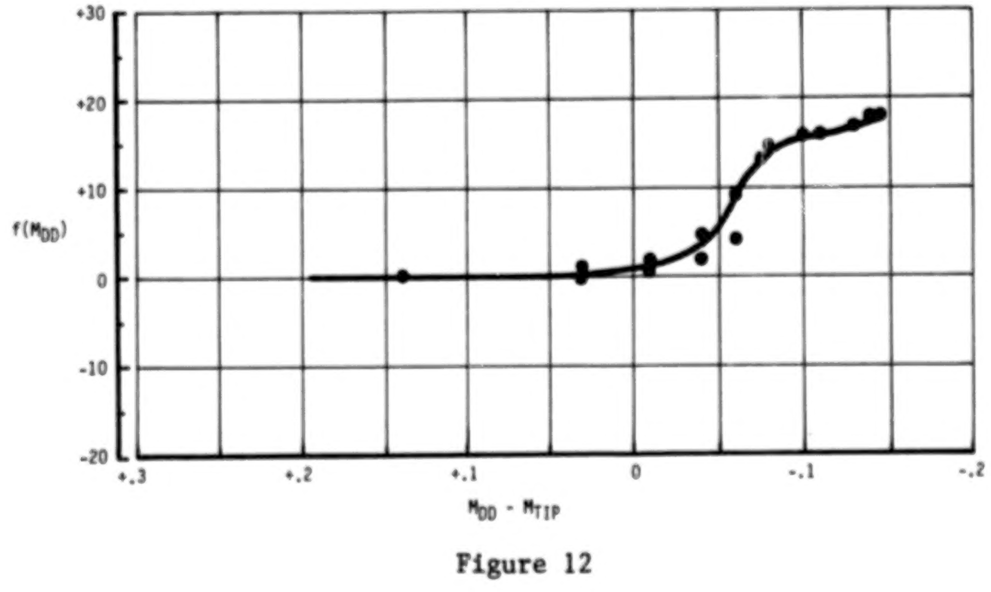

Figure 12

BROADBAND NOISE

To date most broadband noise predictions are based on parametric correlations with test data. In a 1979 NASA Technical Memorandum (ref. 5) several methods were surveyed and supplemented with additional data leading to the formula shown in Figure 13. Even this effort does not give uniformly good prediction accuracy across many configurations and flight conditions. A research effort at Boeing Vertol, which shows a good correlation of overall broadband level with profile power for a given rotor (Figure 14), shows another approach to empirical broadband noise prediction.

$$SPL_{1/3} = 20 \log \frac{V_t^{\,3}}{r} + 10 \log A_b \left(\cos^2 \theta + 0.1\right) + S_{1/3} + f\left(\bar{C}_\ell\right) - 53.3 \quad \text{dB}$$

$$f(C_\ell) = 10 \log \frac{\overline{C_\ell}}{0.4} \qquad (C_\ell \leq 0.48)$$

$$= 0.9 + 80 \log \frac{\overline{C_\ell}}{0.48} \quad (0.48 < \overline{C_\ell} < 0.6)$$

$$f_p = -240 \log T + 0.746\, V_t + 786 \quad \text{Hz}$$

V_t = tip speed

r = distance from source to observer

A_b = total blade area

θ = angle between disc plane and field coordinate

$S_{1/3}$ = 1/3 octave band spectrum for broadband noise

$\bar{C}_\ell$ = average lift coefficient

T = total rotor thrust

f_p = peak frequency

Figure 13

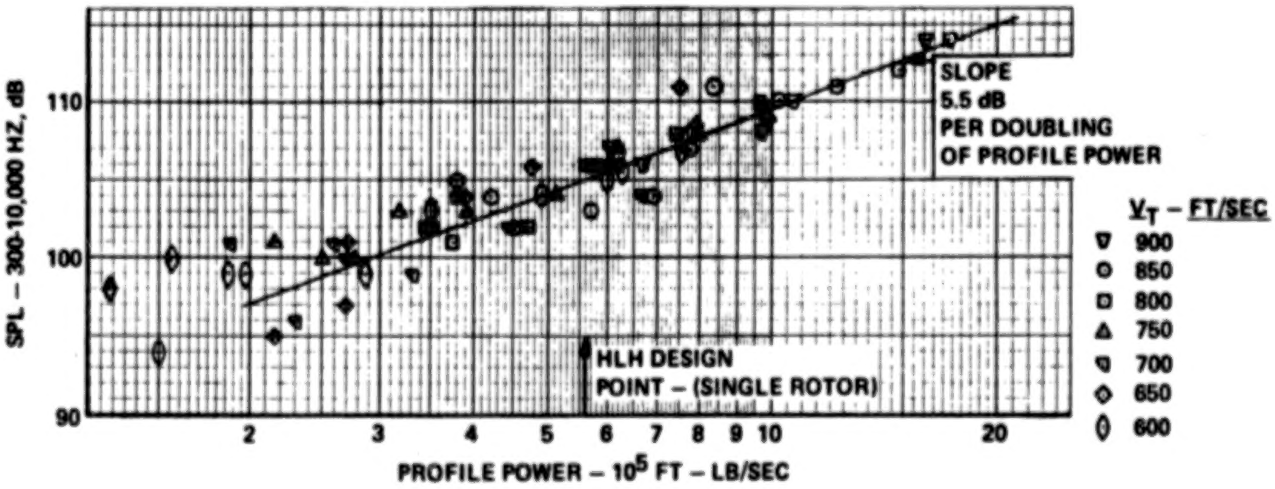

Figure 14

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COMPARISON OF PREDICTED AND MEASURED FLYOVER NOISE

Figure 15 presents comparisons of measured data with predictions using the semi-empirical methodology which has been described. The predictions are done for flyover and approach of two substantially different helicopters, the MBB-BO105, a single rotor helicopter of 5,000 pounds gross weight, and the Boeing Vertol CH-47C, a tandem rotor helicopter weighing 50,000 pounds.

It is not the purpose of this paper to claim that the accuracies depicted are satisfactory or unsatisfactory or that the methods used are superior to other options, but to demonstrate the type of tools currently employed by one manufacturer. As improved empirical or first principle analytical methods become available the appropriate elements of the HELNOP program will be replaced to provide an ever improving capability.

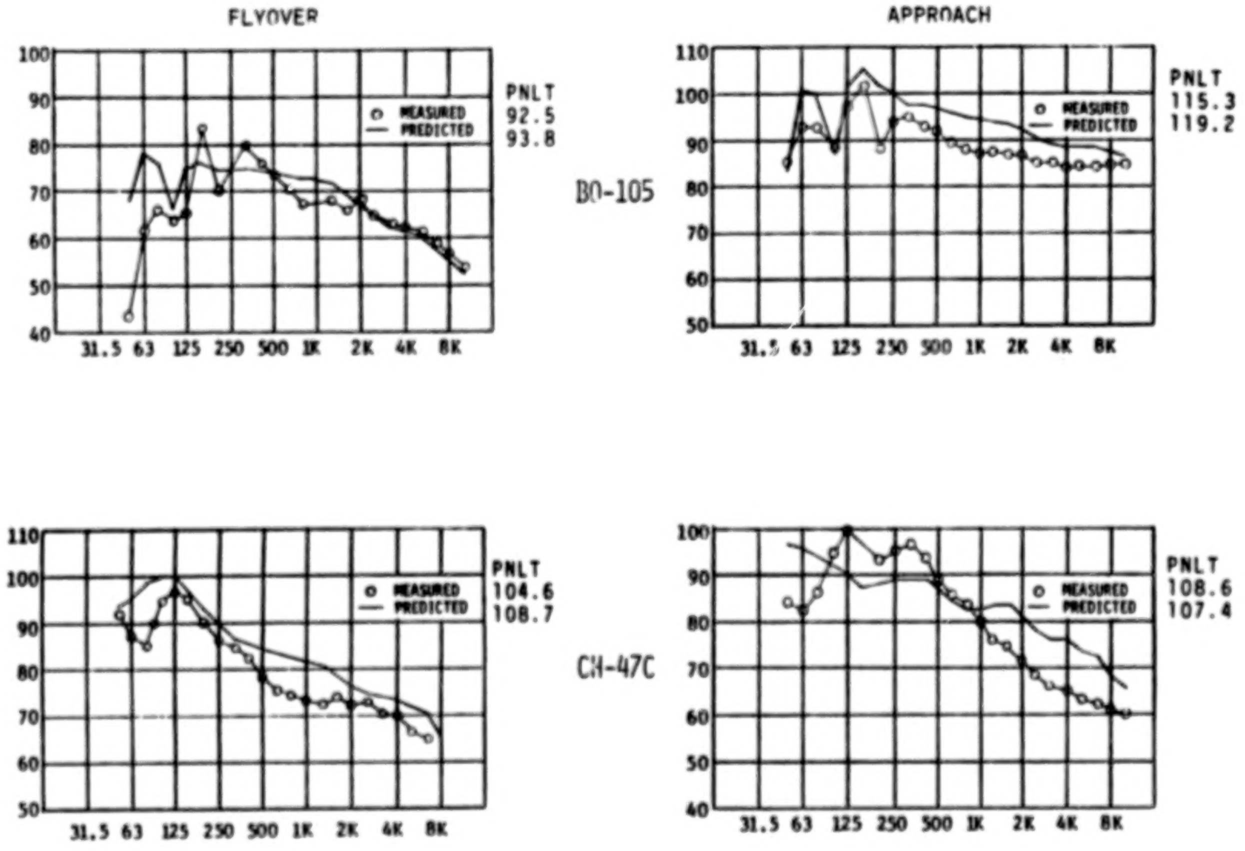

Figure 15

WIND TUNNEL TESTING

Another empirical approach to helicopter noise prediction is through the use of small scale wind tunnel models. Under a recent contract with NASA Ames Research Center the Boeing Vertol Company investigated the correlation of model and full scale test results (Figure 16).

PROGRAM OBJECTIVES

- EVALUATE THE APPLICABILITY OF SMALL SCALE WIND TUNNEL MODELS TO HELICOPTER ROTOR HARMONIC NOISE STUDIES.

BY COMPARISON OF

WIND TUNNEL MODEL AND FULL SCALE FLIGHT TEST ACOUSTICAL MEASUREMENTS

- TRENDS
    - EFFECT OF OPERATING CONDITION
    - EFFECT OF CONFIGURATION CHANGES
- ABSOLUTE VALUES
- USE WIND TUNNEL DATA TO ELUCIDATE INFORMATION ABOUT ROTOR NOISE GENERATION.

Figure 16

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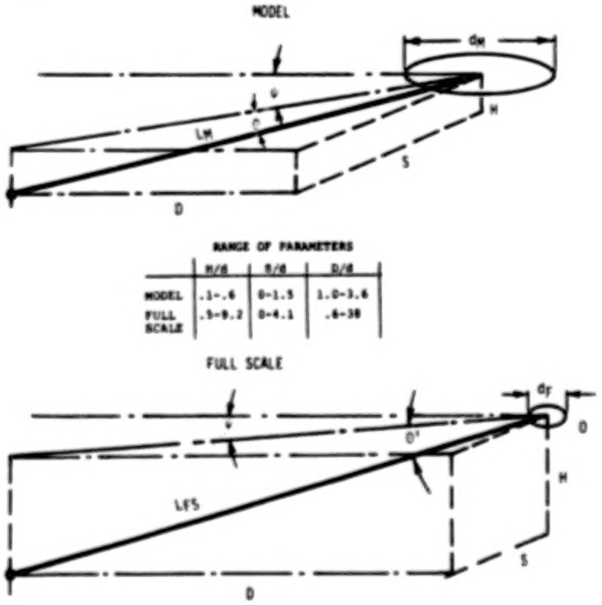
MODEL DATA CORRECTION

The model and full scale data should correspond in directivity to the microphone as closely as possible (Figure 17). This is usually controlled through the selection of full scale aircraft distance D since most other dimensions will tend to be fixed. The other scaling parameters requiring corrections are shown in Figure 18. Reverberation and wind velocity effect corrections are obtained by experimental calibration tests. Note that model and full scale Mach numbers and C_T/σ should be matched.



RANGE OF PARAMETERS

	H/d	S/d	D/d
MODEL	.1-.6	0-1.5	1.0-3.6
FULL SCALE	.5-9.2	0-4.1	.6-38

SELECT FULL SCALE DATA WHEN AIRCRAFT
DISTANCE IS SUCH THAT θ ≈ θ'.

$$\Delta L = 20 \text{ LOG } \frac{L_M/d_M}{L_{FS}/d_{FS}}$$

Figure 17

$$^*SPL_M = SPL_M + \Delta M + \Delta V + \Delta R$$

$^*SPL_M$ = ADJUSTED SOUND PRESSURE LEVEL ~ dB

SPL_M = MEASURED SOUND PRESSURE LEVEL ~ dB

ΔM = ADJUSTMENT FOR MICROPHONE LOCATION ~ dB

ΔV = ADJUSTMENT FOR WIND VELOCITY ~ dB

ΔR = ADJUSTMENT FOR REVERBERATION ~ dB

* MODEL AND FULL SCALE C_T/σ AND TIP MACH NUMBER MATCHED

Figure 18

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HARMONIC DATA COMPARISONS

Figure 19 shows raw model data and the effect of the adjustments for distance and reverberation on an isolated rotor model in hover, while Figure 20 presents the comparison with full scale data for the first twenty harmonics. Figures 21 and 22 present similar data for a tandem rotor helicopter in forward flight.

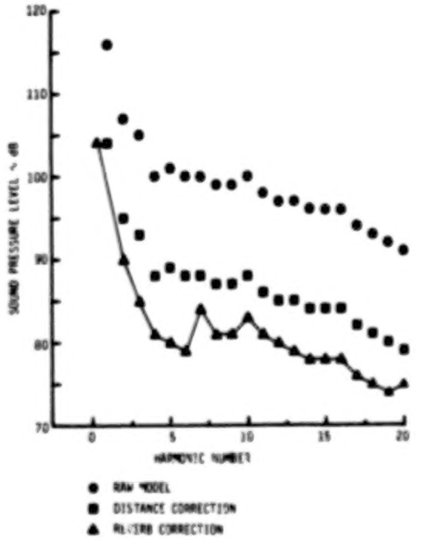

Figure 19

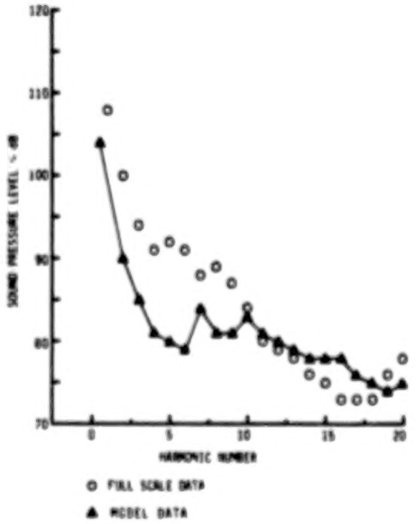

Figure 20

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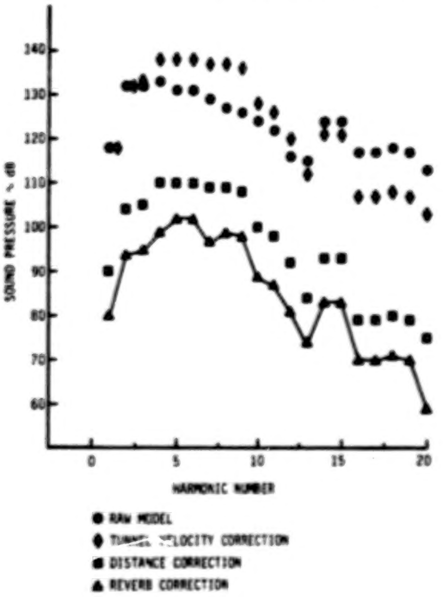

Figure 21

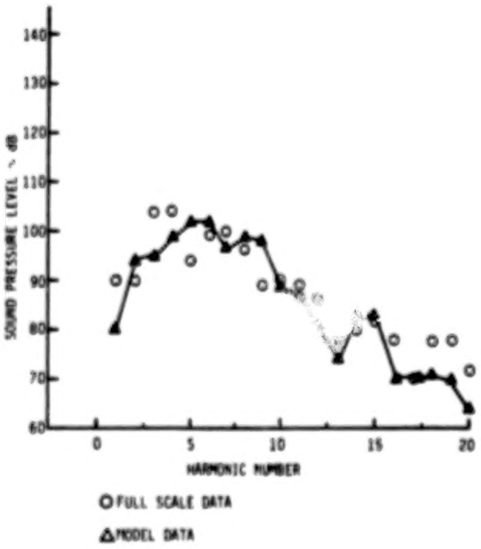

Figure 22

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PEAK TO PEAK DATA

Although harmonic data is of great interest it is difficult to describe or to make comparisons between configurations. Figure 23, which shows a Mach number sweep from model data, clearly indicates the growth of the time domain signal along with the frequency domain data and is amenable to simple evaluation as shown in Figure 24. A reverberation correction is not required in the time domain as long as the microphones are located such that reflected rays do not reach the microphone with a delay time equal to rotor passage period.

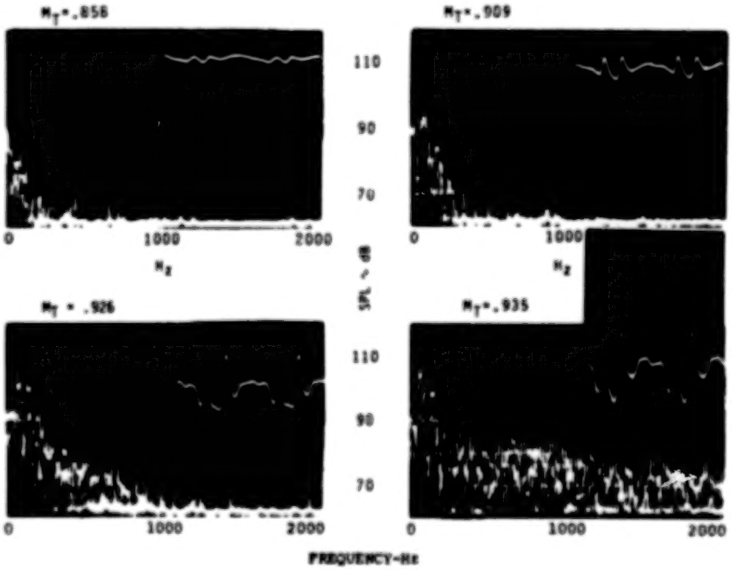

Figure 23

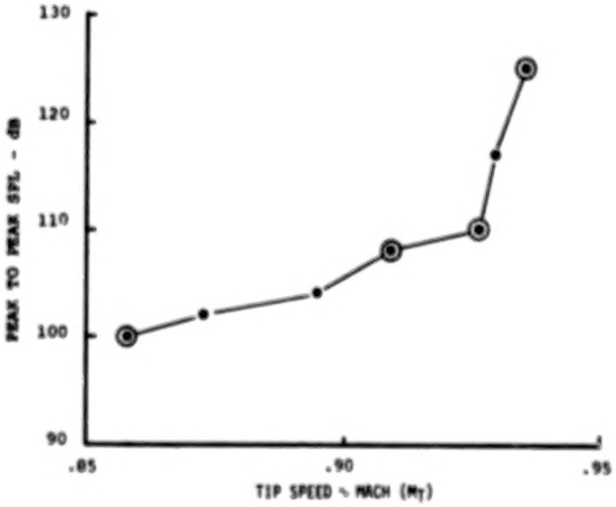

Figure 24

HIGH ADVANCING TIP SPEED

A comparison between adjusted model and full scale peak-to-peak data for a Mach number sweep is presented in Figure 25 while Figure 26 shows the development of highly similar waveforms at high speed.

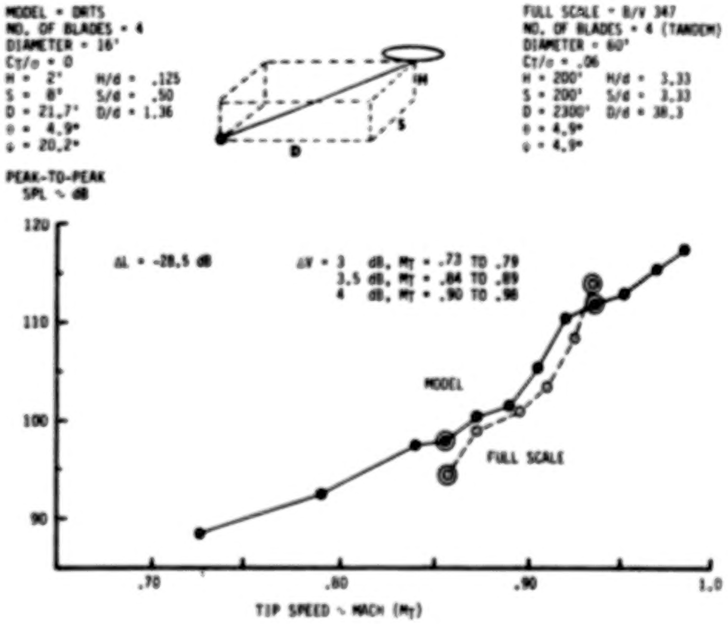

Figure 25

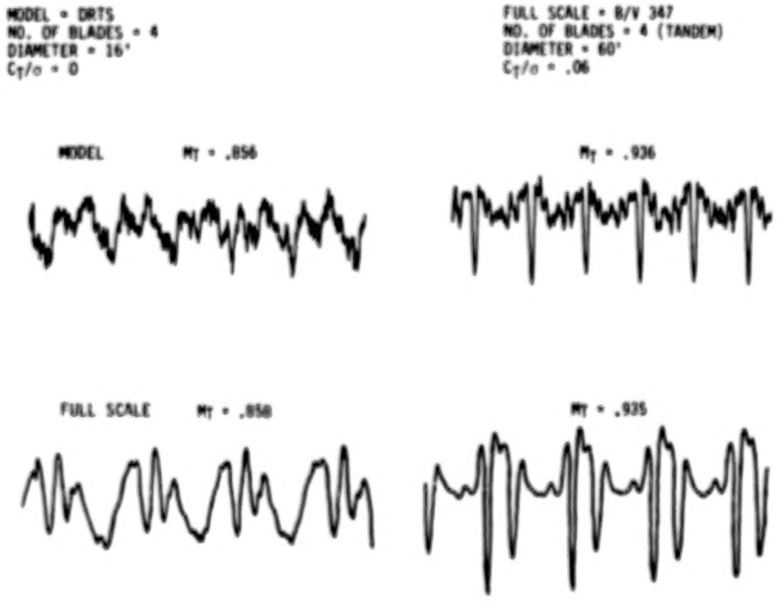

Figure 26

TANDEM ROTOR BLADE-VORTEX INTERACTION

Figures 27 and 28 show the applicability of model testing to predict noise levels due to blade vortex interaction. Note that at low speed (Figure 27) there is relatively little sensitivity to trim (the signals are impulsive at all cyclic trim settings) while at higher speed (Figure 28) the levels of both model and full scale data decrease substantially as rotor separation is increased.

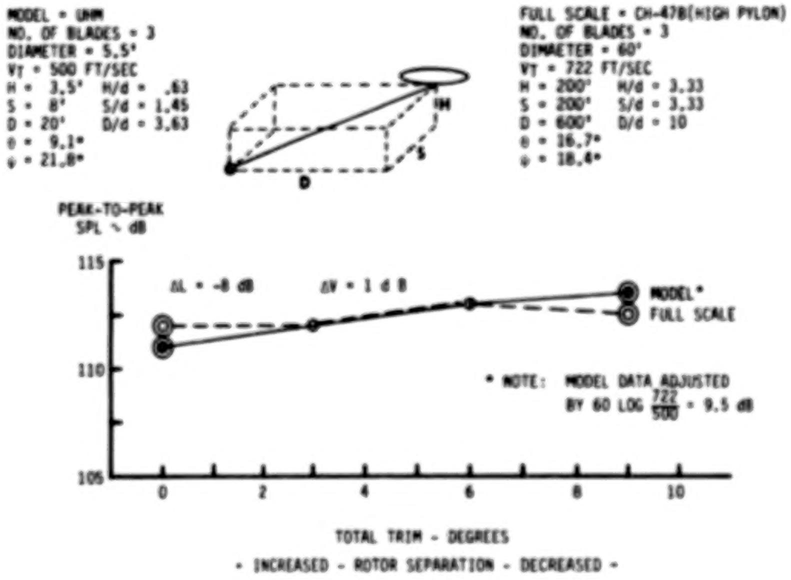

Figure 27

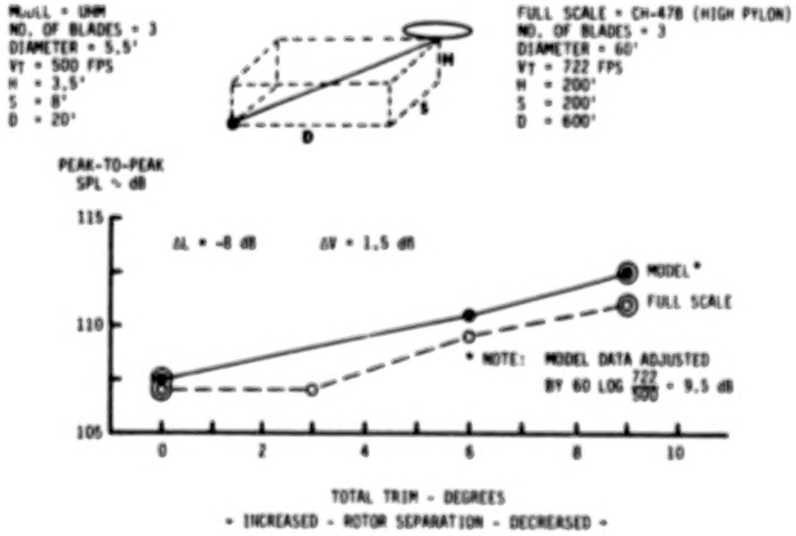

Figure 28

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MAIN-TAIL ROTOR INTERACTION

During testing of the YUH-62A in tied down configuration and I.G.E. hover an impulsive noise at main rotor passage period was radiated behind but not to the sides of the aircraft. Acoustical measurements made during a subsequent wind tunnel test revealed that this phenomenon could be duplicated on the model (Figure 29) and the level predicted fairly closely (Figure 30).

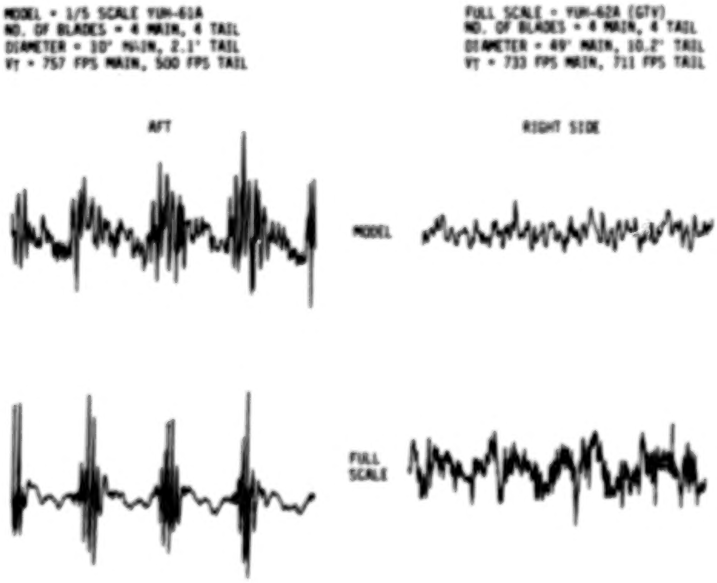

Figure 29

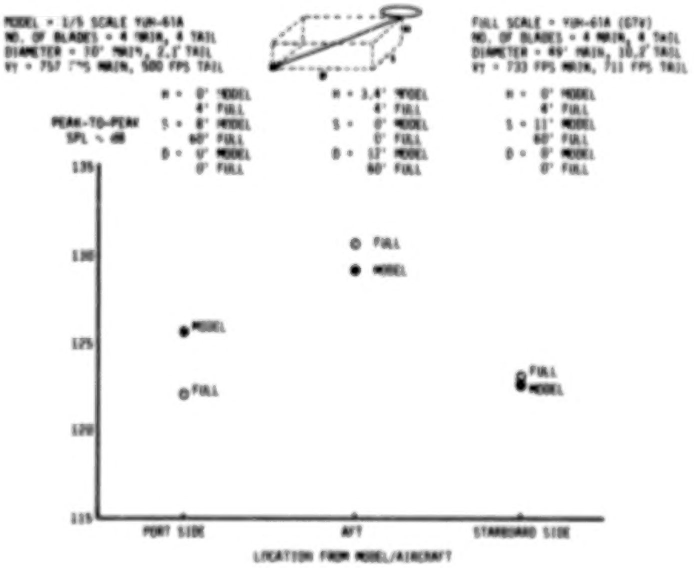

Figure 30

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SUMMARY - WIND TUNNEL MODELS

Figure 31 summarizes all of the comparisons made as part of the investigation. Replication of sound pressure levels from isolated rotors hovering in very low winds was not particularly good due to the highly transient nature of both model and full scale data. For all other conditions the results demonstrate a good potential for the use of performance type wind tunnel models as predictors of full scale acoustical levels.

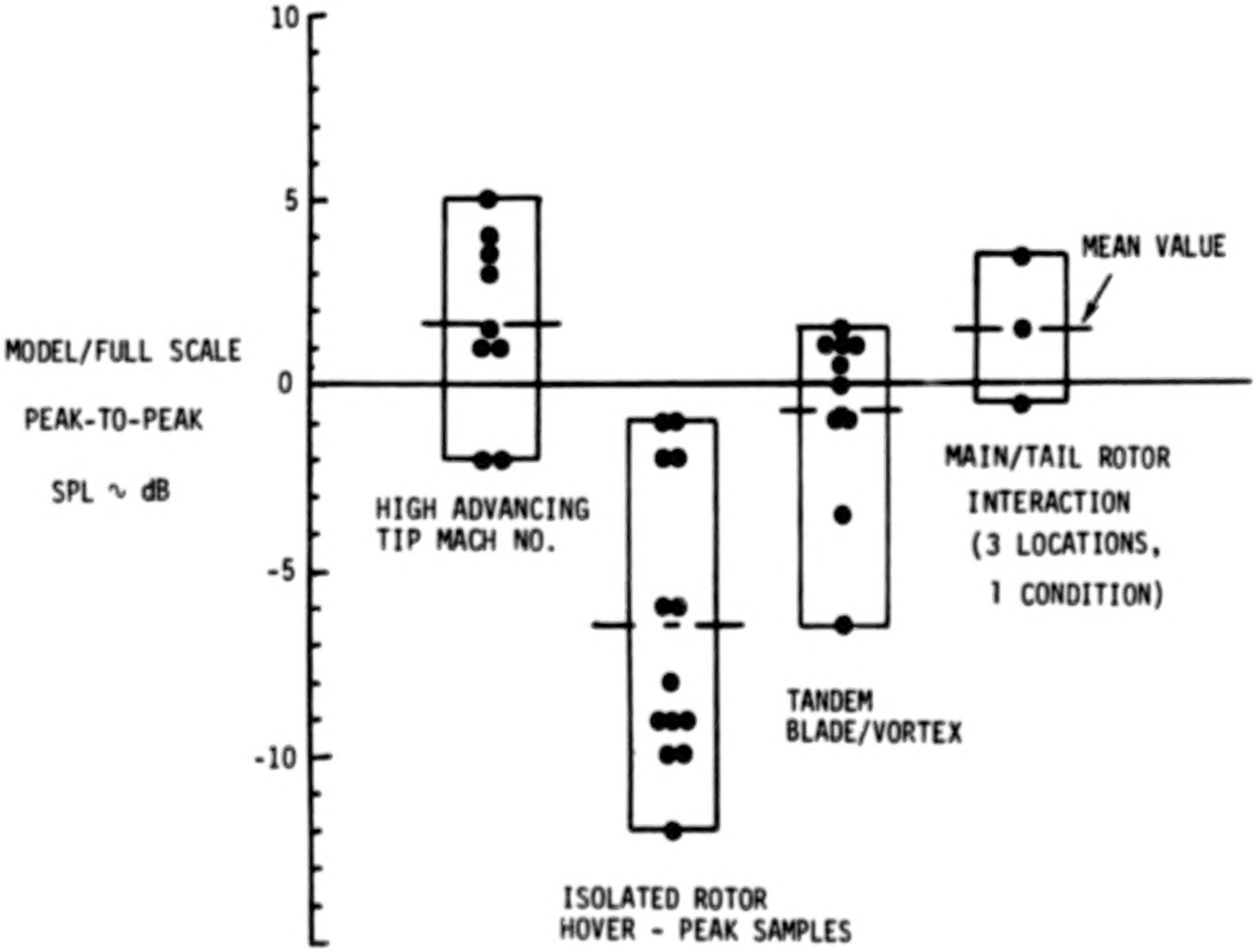

Figure 31

CONCLUDING REMARKS

In industry noise prediction is not simply of academic interest. A significantly inaccurate prediction can have serious impact on helicopter designs, program decisions, and costs. Confidence in a prediction for a new helicopter can only be gained by applying methodology to the most similar aircraft for which data is available and making those adjustments required to arrive at good agreement. At the present such adjustments, whether to input assumptions or final results, are for the most part empirically based and must rely heavily on judgment and experience. Good empirical methodology is not a goal but it is, at present, a necessity.

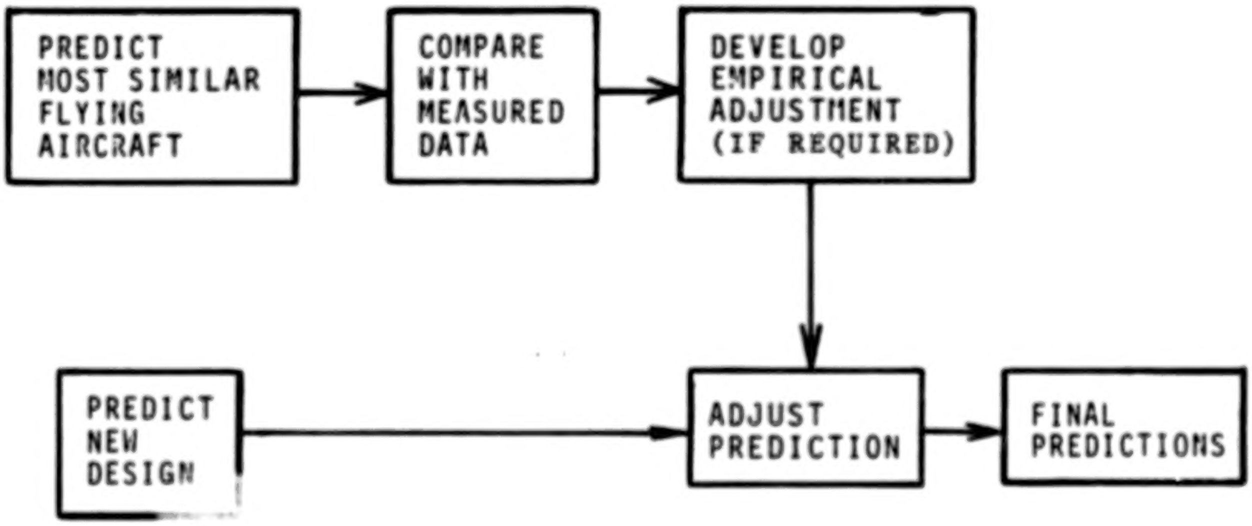
PREDICT MOST SIMILAR FLYING AIRCRAFT
COMPARE WITH MEASURED DATA
DEVELOP EMPIRICAL ADJUSTMENT (IF REQUIRED)
PREDICT NEW DESIGN
ADJUST PREDICTION
FINAL PREDICTIONS

Figure 32

REFERENCES

1. Lowson, M. V. and Ollerhead, J. B.: Studies of Helicopter Rotor Noise. USAAVLABS Technical Report 68-60, 1969.

2. Wright, S. E.: Discrete Radiation from Rotating Periodic Sources. J. Sound & Vib., vol. 17, no. 4, pp. 437-498, Aug. 22, 1971.

3. Schmitz, F. H. and Yu, Y. H.: Transonic Rotor Noise - Theoretical and Experimental Comparisons. Paper No. 22, Sixth European Rotorcraft and Powered Lift Aircraft Forum, 1980.

4. Sternfeld, H. and Wiedersum, C.: Study of Design Constraints on Helicopter Noise. NASA CR-159118, 1979.

5. Pegg, R. J.: A Summary and Evaluation of Semi-Empirical Methods for the Prediction of Helicopter Rotor Noise. NASA TM-80200, 1979.

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PRIORITY FOR EMPIRICAL METHODS DEVELOPMENT

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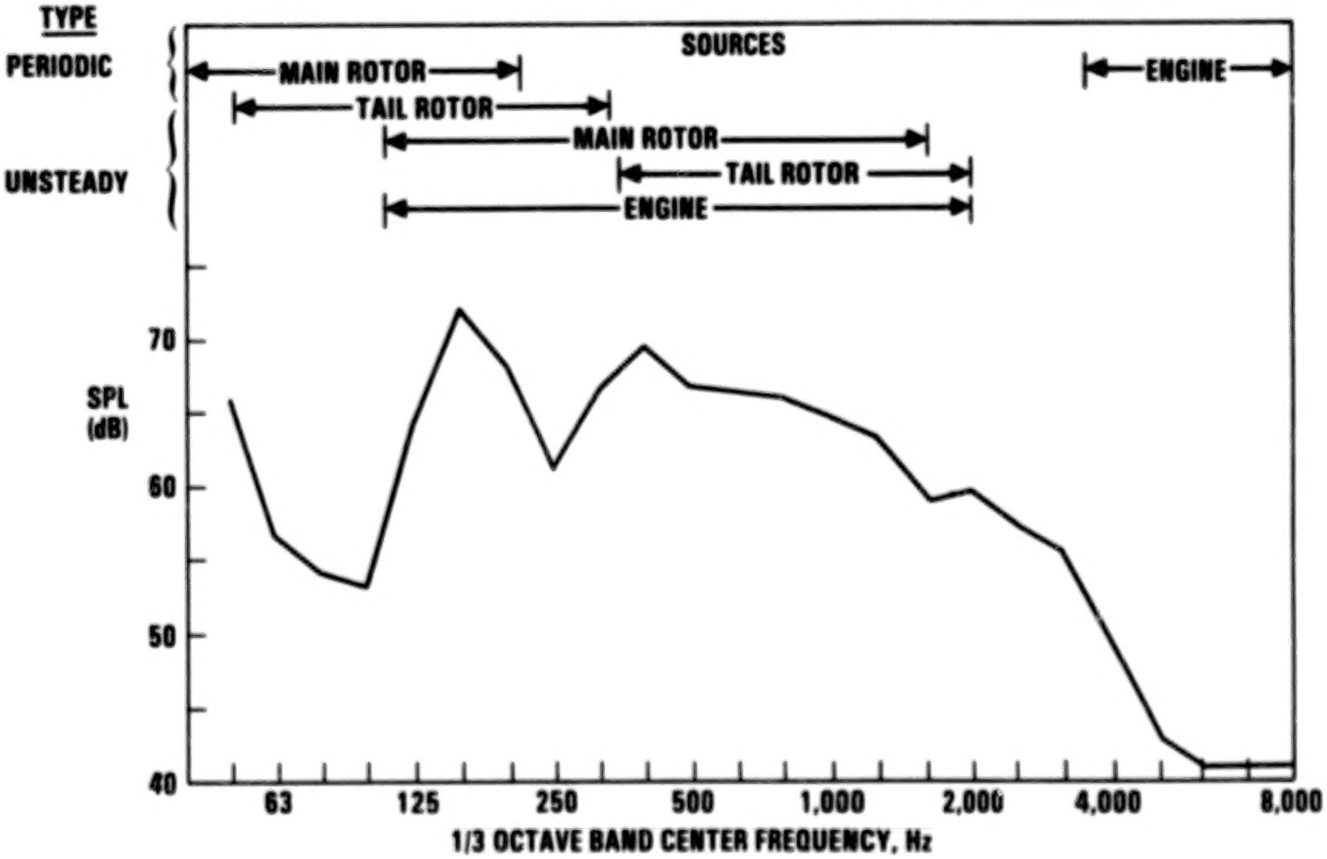
HELICOPTER NOISE SOURCES BY FREQUENCY RANGE

Several noise sources combine to make up the total helicopter noise spectrum shown in one-third octave bands in the figure. For this particular helicopter, a Hughes 500D operating in cruise, main rotor discrete frequency noise is of little consequence. The sources that are most important to community annoyance are the tail rotor (discrete), main rotor (unsteady) and engine (unsteady). It is clear that all three main helicopter noise sources must be controlled if community noise abatement is to be accomplished. For the approach condition and for many helicopter models in high speed level flight, main rotor impulsive noise becomes a fourth important source.



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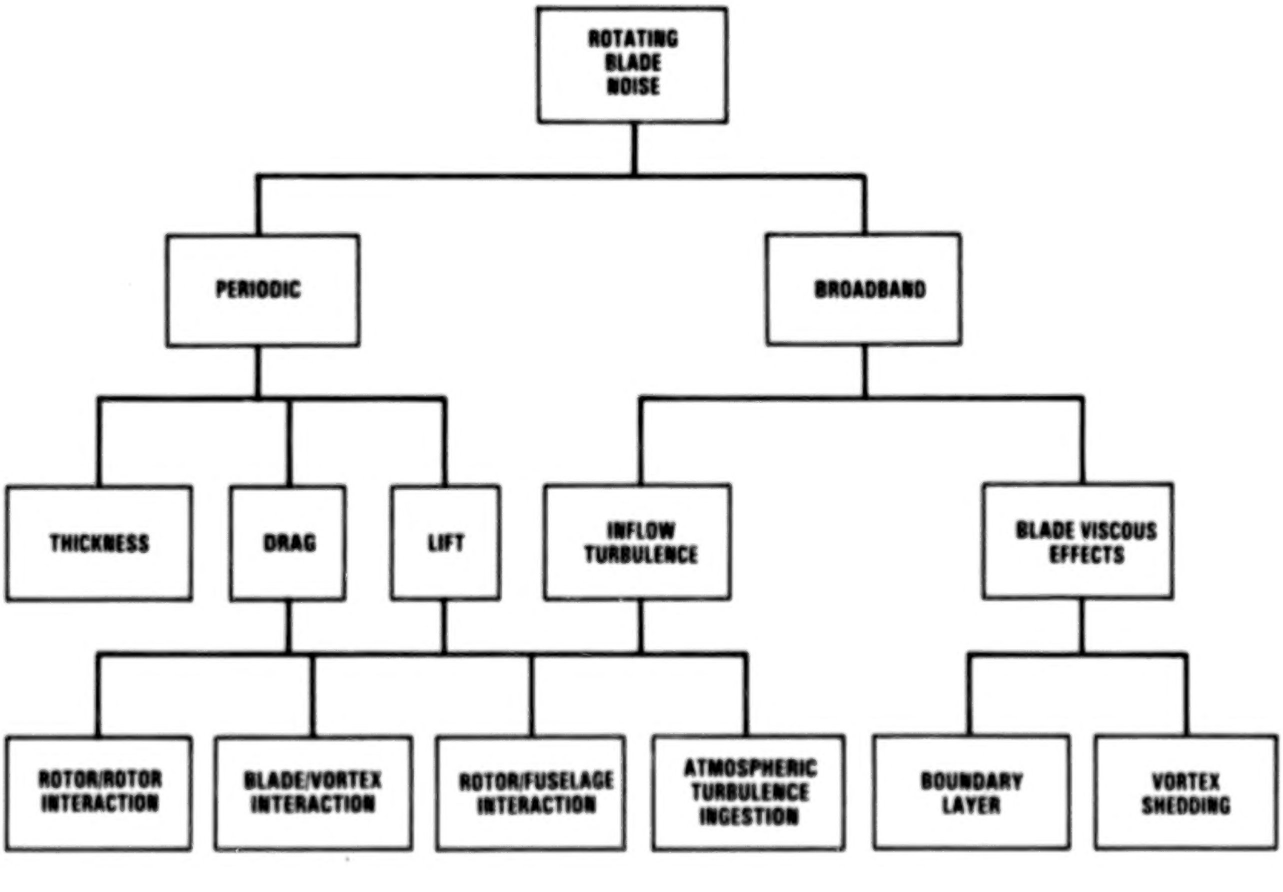
ROTOR NOISE COMPONENTS

Many components, generally separated into periodic and broadband types, combine to make up the whole of the rotor noise spectrum. This figure is similar to many existing ones showing the relationship between types of noise and sources, but it is slightly different. Note the linkage between the various types of interaction and turbulence induced sources and the next level on the chart. The intent is to show the interrelationship between periodic and broadband mechanisms in rotor noise without the categorization normally indicated. It is not possible to make a clear distinction between the various types and sources of noise produced by a rotor, but lack of this insight should not be an obstacle to generation of useful models for prediction of the noise results.


ROTATING BLADE NOISE
PERIODIC
BROADBAND
THICKNESS
DRAG
LIFT
INFLOW TURBULENCE
BLADE VISCOUS EFFECTS
ROTOR/ROTOR INTERACTION
BLADE/VORTEX INTERACTION
ROTOR/FUSELAGE INTERACTION
ATMOSPHERIC TURBULENCE INGESTION
BOUNDARY LAYER
VORTEX SHEDDING

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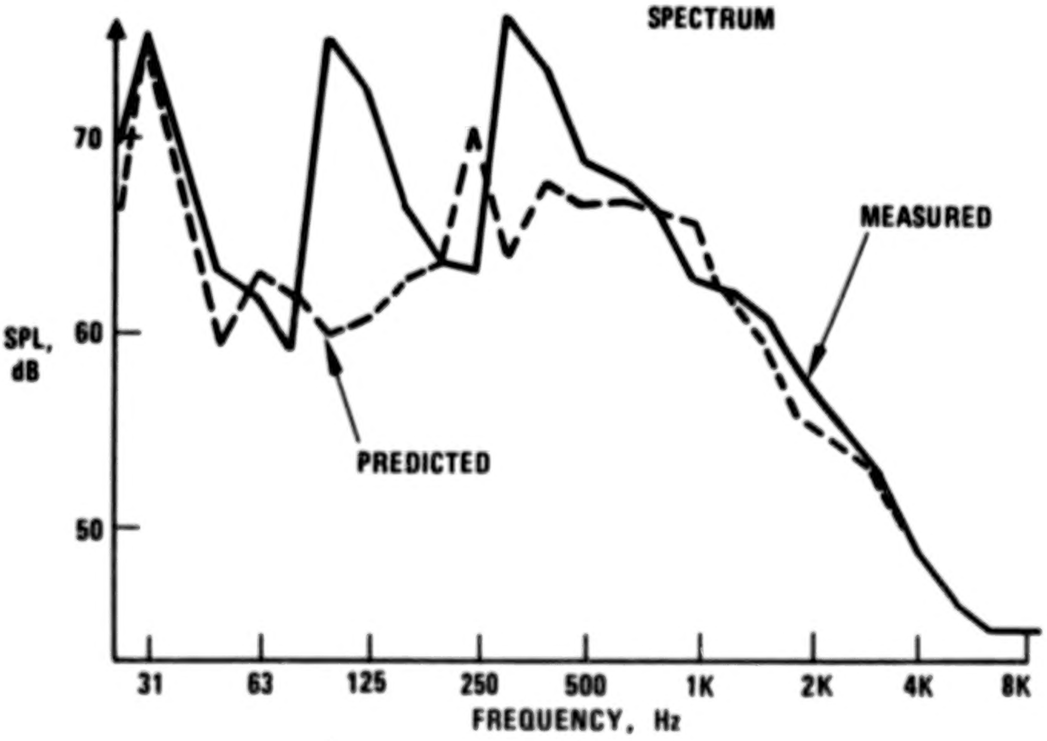
PREDICTION ERROR IS DIRECTLY PROPORTIONAL TO DETAIL ATTEMPTED

Even though current methods of estimating the gross noise levels of helicopters are not very accurate, they are more accurate than some of the more detailed methods are in predicting spectral detail. Spectrum detail is necessary for application of noise control to the helicopter and lack of accuracy can be very costly in terms of ineffective results produced and the number of iterations required to define the final solution. The three-position EPNL prediction is within 1.3 EPNdB in the example shown in the figure. The error grows to 1.6 and 3.3 EPNdB and PNdB for the single position time-integrated unit and maximum tone corrected noise level respectively. Obviously there were compensating errors in the data (not uncommon, but also not consistent) that become unavailable as the level of detail increases. Finally, the detailed spectrum estimate shows errors of over 10 dB due to the fact that one source (the tail rotor) was grossly underestimated. The tail rotor would not have been considered a very significant source if the predicted spectrum were the only source of information and the design of the helicopter may have continued until the test data disclosed the bad news. This type of prediction performance can not be afforded by the industry.

	3-POSITION EPNL	CENTERLINE EPNL	CENTERLINE PNLTMAX
MEASURED	88.3	88.7	89.4
PREDICTED	87.0	89.1	86.1
ERROR	1.3	1.6	3.3

SPECTRUM



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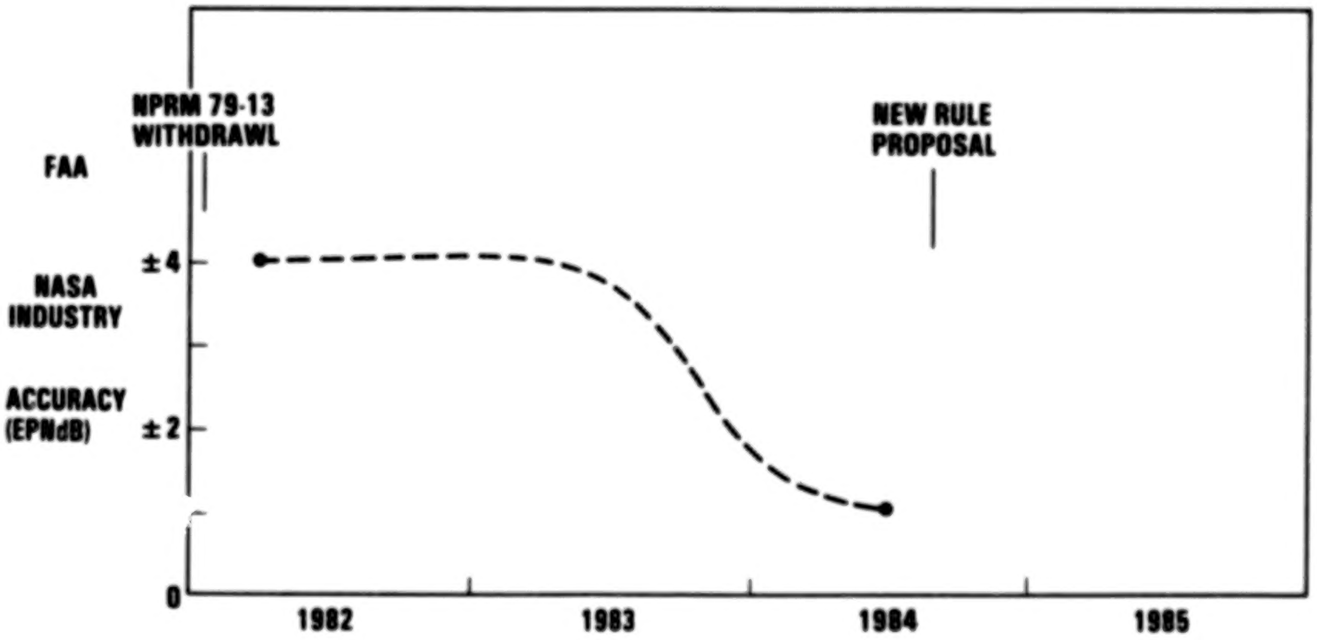
NOISE PREDICTION DEVELOPMENT TIMETABLE

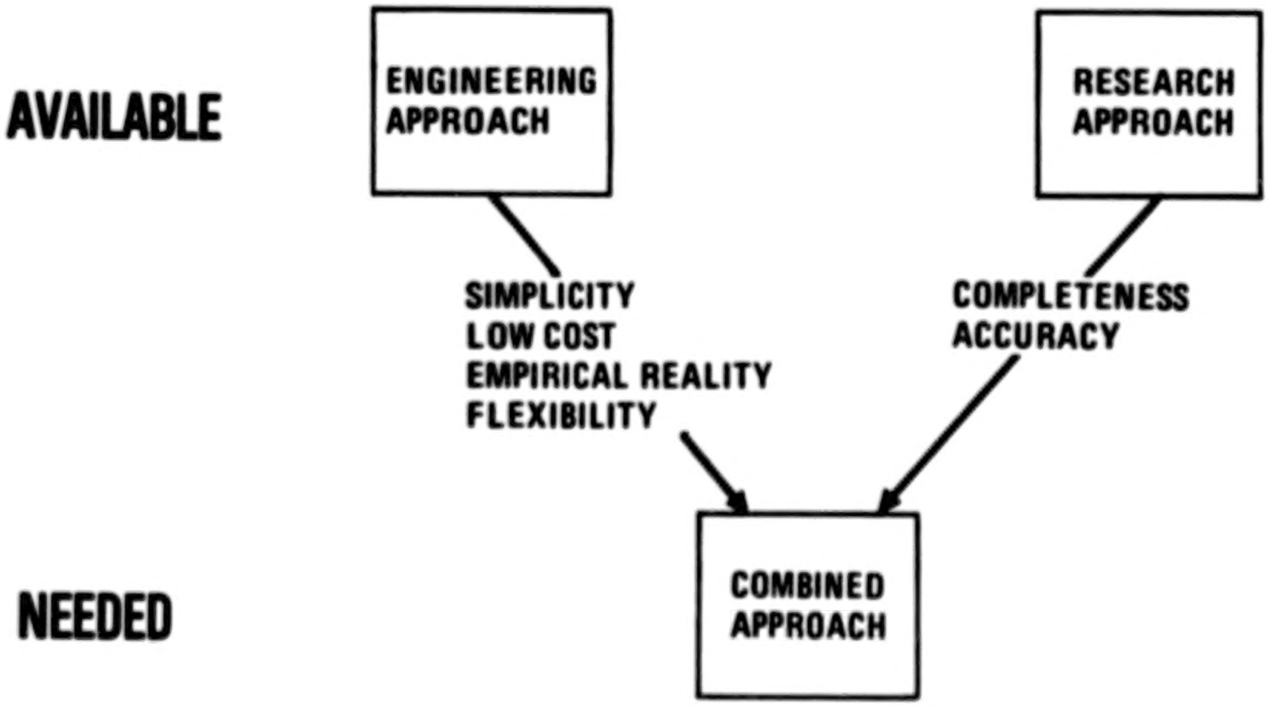
Current helicopter noise prediction accuracy for high confidence requires on the order of ± 4 EPNdB margin for design of new helicopters. It has been acknowledged that such a margin forces the manufacturers of new helicopters into designs that are much less cost effective than those without a noise constraint. Industry is committed to the FAA to come forth with a new certification rule proposal by 1984. It is necessary to have cut the required design margin substantially by that time in order to propose a rule which is both economically acceptable to the helicopter industry and supportive of environmental goals. There is pressure for the industry and NASA to improve the existing rotorcraft noise prediction situation in the short term.



APPROACH TO ROTORCRAFT NOISE PREDICTION

Both the engineering (semi-empirical) and research (first principles) approaches to rotorcraft noise reduction have their advantages. The research approach has the advantages of completeness and better understanding of the physical mechanisms that generate the noise. However, many questions arise regarding the availability and utility of such methods. It is clear that the first principles analysis methods require a good deal more detail in input to the acoustic codes. This detail necessitates a large quantity of information generation prior to the actual acoustic calculation. Cost and time are large issues in this type of approach. A second problem is that of availability time frame. As stated earlier, the prediction procedures must be available in a timely manner relative to anticipated regulatory requirements. It is felt that there is no alternative in the near term to well validated semi-empirical analyses.

TYPES OF PREDICTION NOW PERFORMED


AVAILABLE
ENGINEERING APPROACH
RESEARCH APPROACH
SIMPLICITY
LOW COST
EMPIRICAL REALITY
FLEXIBILITY
COMPLETENESS
ACCURACY
NEEDED
COMBINED APPROACH

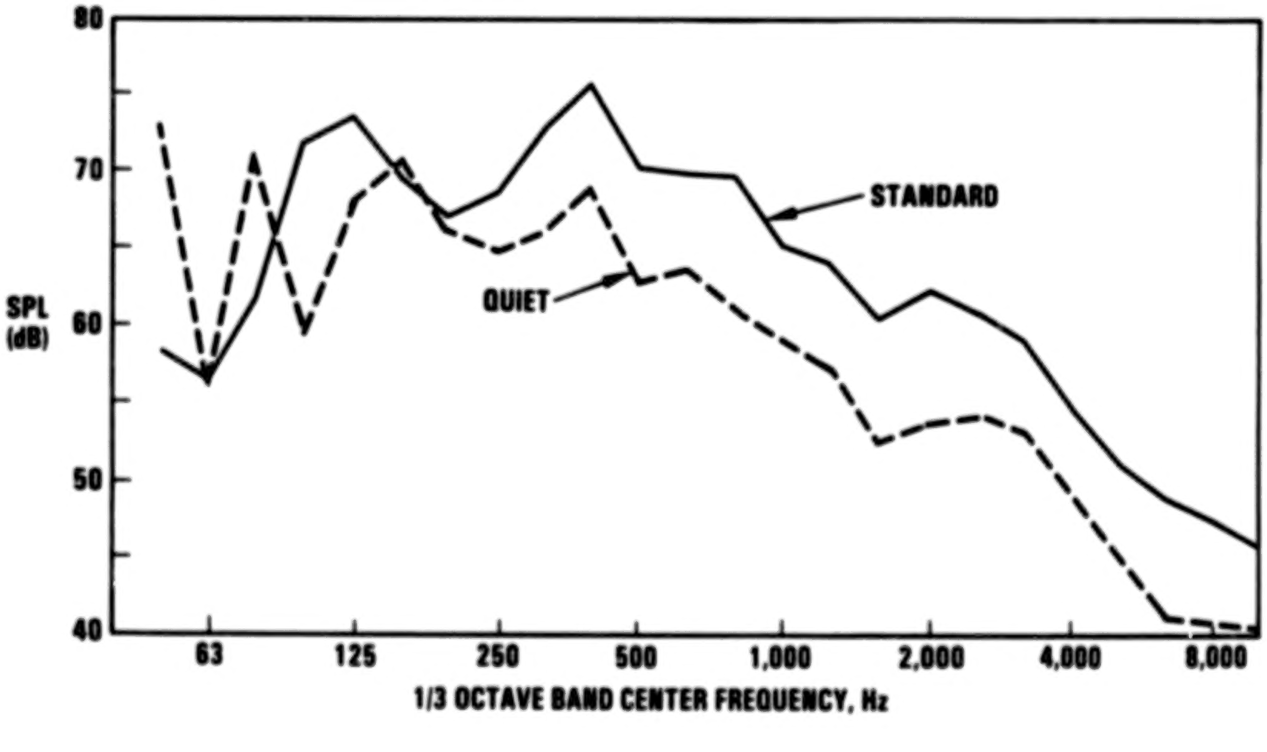
HELICOPTER NOISE SOURCE RANKING

It is not possible to rank the relative importance of helicopter noise sources in general. It varies by model and flight condition. The ranking shown in the figure is for a small single multi-bladed main rotor helicopter. The tail rotor is the primary source for two of the three flight conditions listed making it the major problem on this helicopter. However, if the tail rotor can be quieted it is obvious that the main rotor is the controlling source in all three flight regimes. Further, the broadband portion of the main rotor spectrum is the most important source since it appears in all three flight conditions. Since tail rotor noise can be controlled at much lower expense than main rotor noise (to be demonstrated later), it can be projected that the main rotor, particularly the broadband portion of its spectrum, will be the main source for the helicopter in almost all cases. Attention for both prediction and control must be directed primarily at this source.

FLIGHT CONDITION	**SOURCE**			
	RANK ORDER **RANK ORDER**	**1**	**2**	**3**
LANDING	**1**	**MAIN ROTOR**	**TAIL ROTOR**	**ENGINE**
FLYOVER	**2**	**TAIL ROTOR**	**MAIN ROTOR**	**MAIN ROTOR**
TAKEOFF	**3**	**TAIL ROTOR**	**MAIN ROTOR**	**ENGINE**

QUIET TAIL ROTOR EFFECTIVENESS IN CRUISE

The difference in H500D overflight noise in cruise made by changing from a standard to a quiet tail rotor is significant, as shown in the figure. The audible difference is even more remarkable. With the quiet tail rotor installation, the total noise character of the helicopter changes from a tail rotor dominated buzz to a smooth broadband sound. The change is effected by exchanging the standard two-bladed, 703 fps tail rotor with a four bladed, staggered, 530 fps unit. The difference in empty weight for this installation is only on the order of 9 lb and performance change is negligable. The point is that tail rotor noise can be controlled with little impact to the overall helicopter system.



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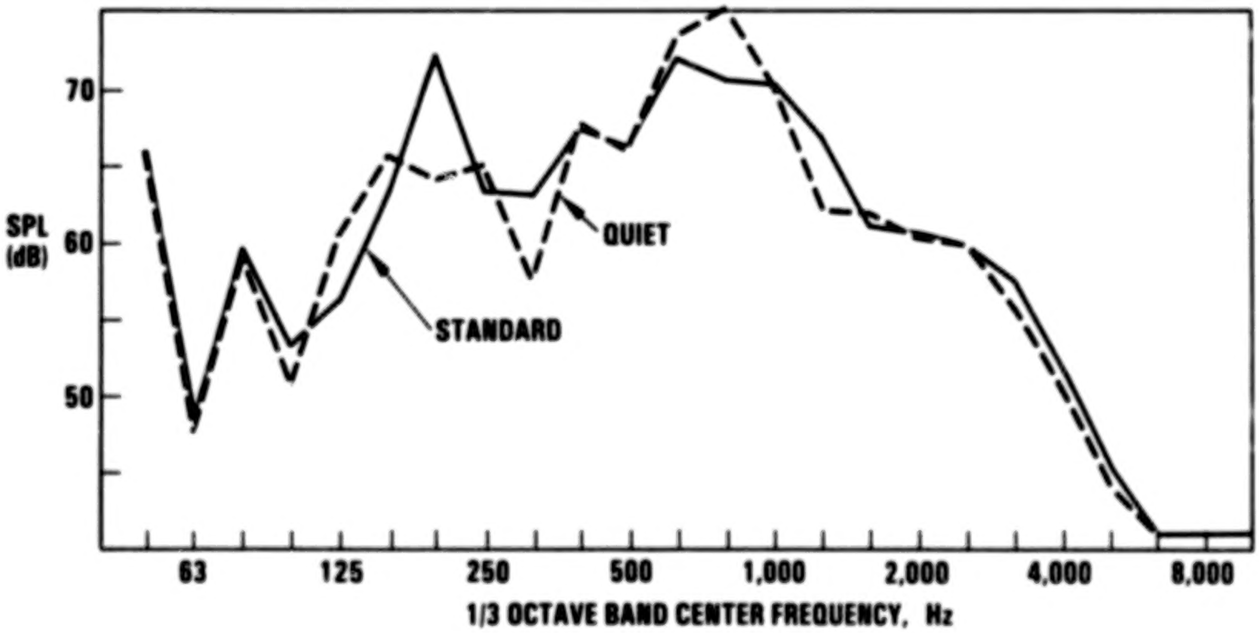
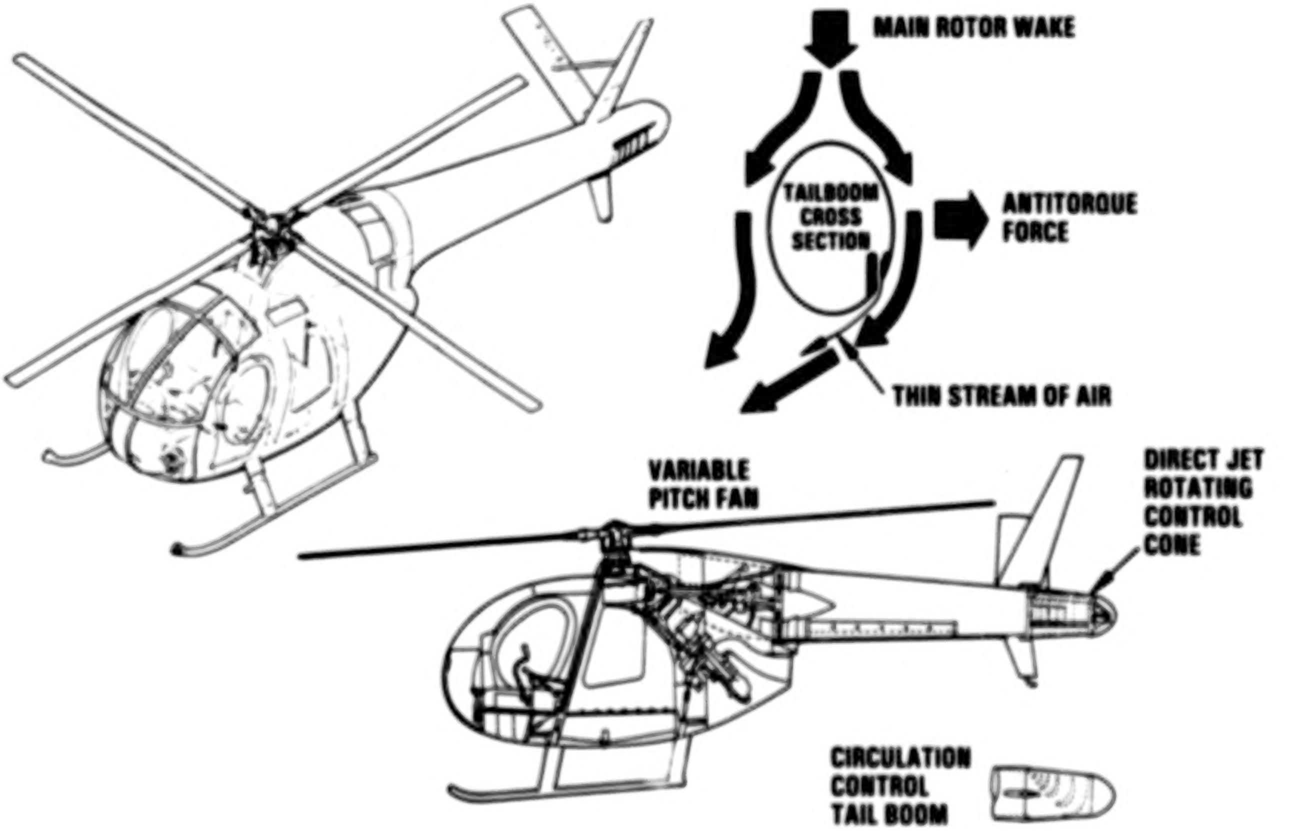
QUIET TAIL ROTOR IMPROVEMENT IN APPROACH

The quiet tail rotor helps the noise situation of the H500D substantially in cruise and takeoff but not in approach, as shown in the figure. The reason for this is that the tail rotor does not contribute a good deal of noise to the total in this particular flight regime and as a consequence the total noise is not reduced when that component is modified. In this flight condition the main rotor is unquestionably the primary noise source and must be worked on.



THE NOTAR ANTITORQUE CONCEPT FOR QUIET

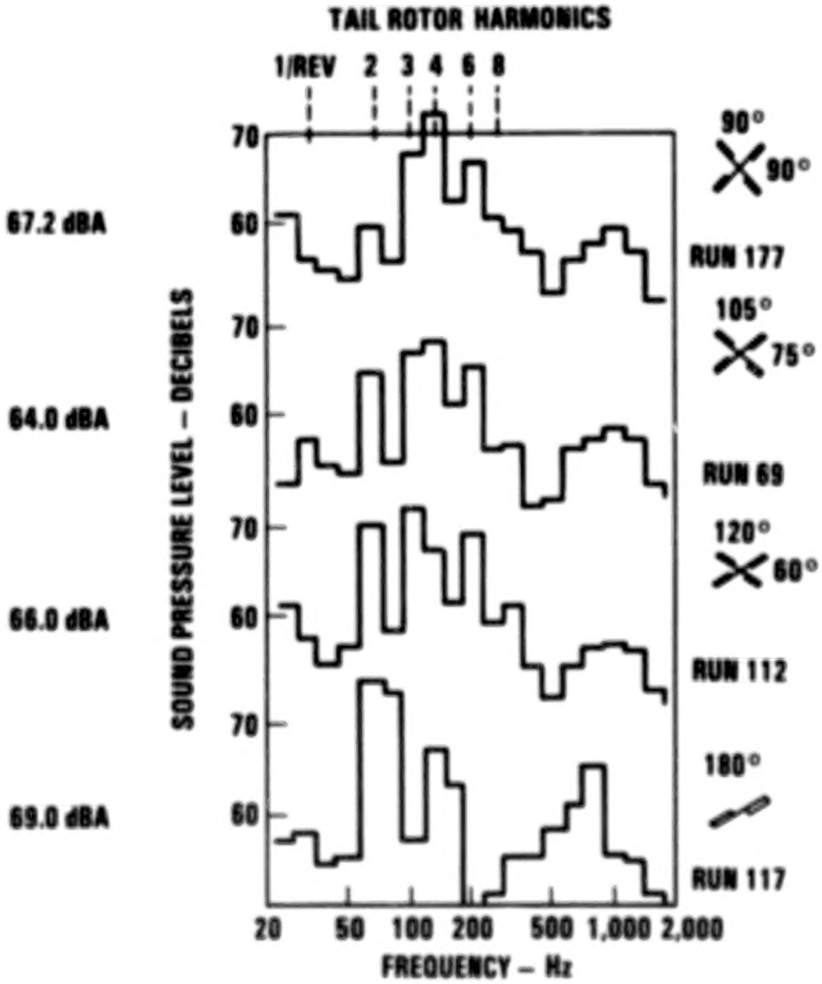
The concept of NOTAR (NO TAil Rotor) is shown in the figure. Thrust to counter main rotor torque is generated by a combination of circulation control lift on the tail boom and direct thrust from a jet. A single stage fan mounted near the root of the tail cone supplies flow to both the circulation control slot and the thruster nozzle. Flow between the two outlets is apportioned according to the flight condition. Acoustic evaluation of the prototype NOTAR vehicle will be performed in the near future. This is potentially a low noise system because of the location of the fan and the low velocities used to generate lift.


MAIN ROTOR WAKE
TAILBOOM CROSS SECTION
ANTITORQUE FORCE
THIN STREAM OF AIR
VARIABLE PITCH FAN
DIRECT JET ROTATING CONTROL CONE
CIRCULATION CONTROL TAIL BOOM

TAIL ROTOR NOISE REDUCTION BY STAGGER AND TAIL ROTORS IN GENERAL

The figure below indicates the noise reductions available from a low speed tail rotor through the use of blade number and blade stagger. The bottom one-third octave band spectrum is for a two bladed tail rotor which produces 69 dBA with multiples of 2/rev frequencies predominating in the rotational portion of the spectrum. A conventional four bladed tail rotor with even blade spacing at the same lift and speed is shown at the top of the figure. The average lift coefficient is reduced by a factor of two relative to the two-blader, the dominant rotational noise is at 4/rev, and the noise level has been reduced approximately two dBA. The two center plots show varying spectral distribution between 2 and 4/rev harmonics and lower noise levels than the other cases. The lowest noise setting investigated was the 75/105 degree stagger case with a net noise reduction of 5 dBA relative to the original two-bladed configuration. This change coupled with the 26 percent tip speed reduction give the quiet tail rotor its markedly different noise performance.

Having shown that, at least in one instance, tail rotor noise can be reduced substantially with relatively inexpensive changes, and that alternate quiet antitorque systems will become available in the next helicopter generation, we have concluded that tail rotor noise is a second order priority task. Primary R&D work should be aimed at the main rotor.



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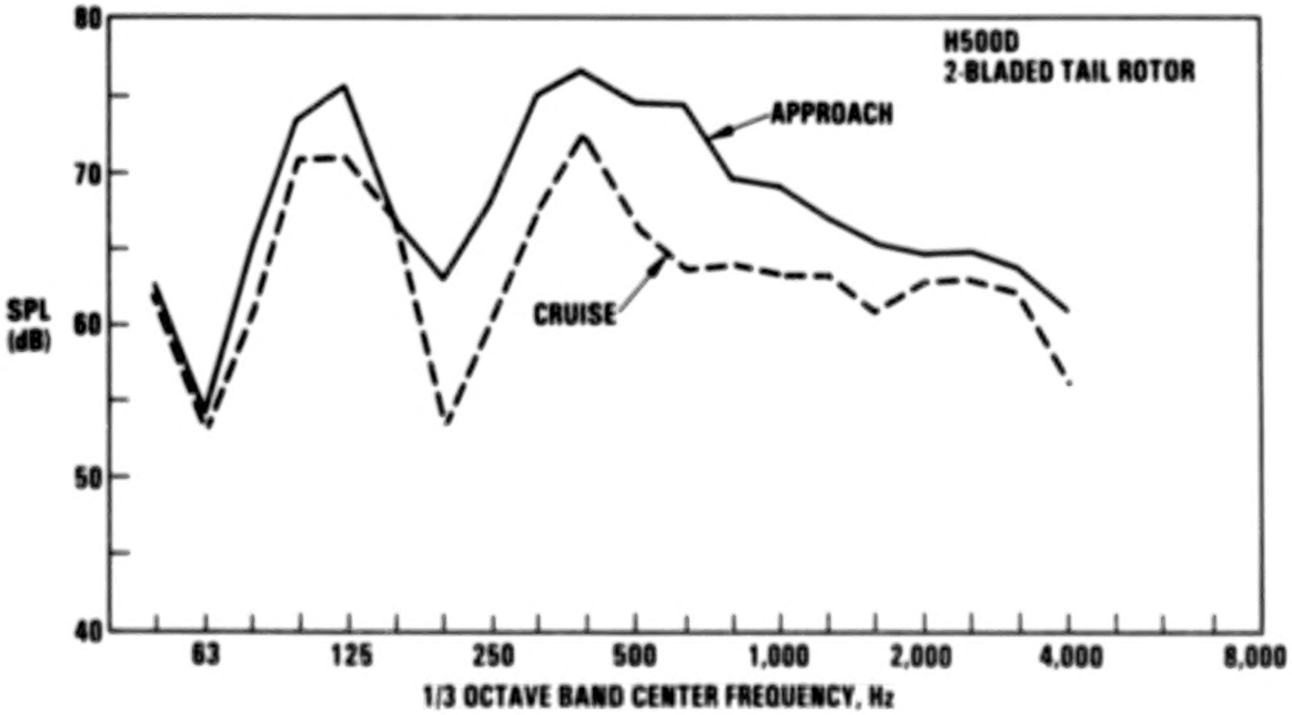
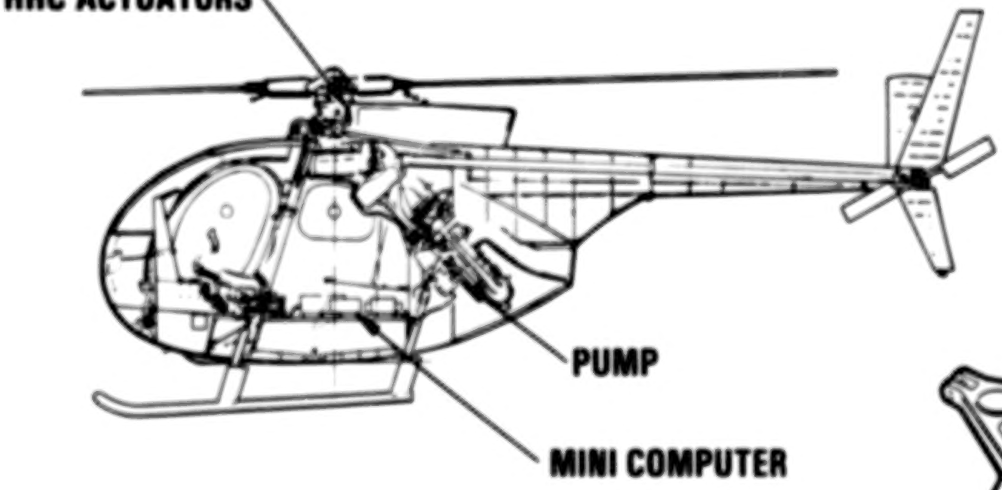
BLADE/VORTEX INTERACTION NOISE - DESCENDING FLIGHT

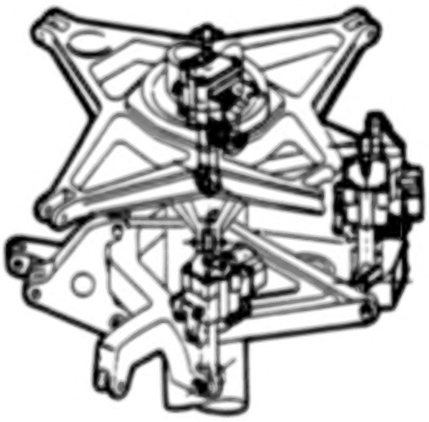
The approach flight condition is a critical one for most helicopter models because of a combination of higher source levels and closer proximity to the community necessitated by common glideslopes. The two noise spectra in the figure are for the same helicopter at 60 knots in level flight and in cruise. The only difference between the conditions is the six degree descent glideslope for the higher noise level case. Main rotor blade vortex interaction is the reason for the higher noise levels. This source must be controlled, but it does not seem to lend itself to empirical methods of prediction (and hence control). Here there is no doubt that a more analytical approach is required to solve the problem. The location, strength, and distribution of the shed tip vortices must be forecast and their interaction with succeeding blade passages and the resulting acoustic effects predicted. Since these events vary so widely between helicopter type and flight condition there can be very little generalization regarding the noise produced. Some well planned studies are underway in this technical area and they should be carried through to practical application as soon as possible.



HIGHER HARMONIC CONTROL

The Higher Harmonic Control (HHC) system, soon to be flight tested by Hughes Helicopters, has potential for controlling the Blade-Vortex Interaction noise problem. HHC applies high frequency feathering to the main rotor blades to alleviate loads generated aerodynamically on the blades. Since BVI noise is the result of interaction between a blade and a shed tip wake, the loads (and hence acoustic effects) caused by such intersections could presumably be reduced by appropriate control inputs. This method of alleviation could be used as a supplement to other methods of control considered to date such as tip shape changes and tip speed reduction.





NOISE PREDICTION - TOOLS AND NEEDS

In addition to several simple trending methods for "ball parking" helicopter noise, HHI uses two basic types of noise prediction method, as indicated in the figure. Preliminary noise spectrum estimates are made with the program HEXNOP which is a frequency based method for main and tail rotors with an engine noise routine. Rotational noise is computed using the Lowson-Ollerhead method with empirically derived harmonic loading constants based on experience with our helicopters. Broadband, BVI, and thickness noise are predicted with predominantly empirical methods using only gross rotor parameters and operating conditions as inputs. The method is quick and inexpensive to use and yields reasonable accuracy for helicopters of conventional design. The accuracy is not good enough to make this program entirely suitable for "specification" type noise predictions.

The RAPP method is used for more critical and detailed helicopter noise studies and in cases where noise-time history information is necessary. It is time-based, uses the Lowson compact source theory for rotational noise, and uses the Army-Ames developed linear code for thickness noise. Broadband noise is predicted with a semi-empirical model of an oscillating lift dipole with section related Strouhal frequencies. This model is used for convenience only since it is recognized that this is not the actual mechanism associated with this noise. Since this method takes into account the detailed aerodynamics of the rotor system it is capable of more accuracy and accounts for many more parameters than the frequency based method.

Current prediction needs are seen to be in the areas of validated broadband noise prediction and transient rotor airload (Blade-Vortex interaction) noise. Needs for future generation helicopters will include high speed thickness noise and more detailed rotational noise prediction using noncompact source models.

- Noise prediction tools
  - Preliminary estimates - HEXNOP
  - Detailed studies - RAPP
- Noise prediction needs
  - Now
    - Validated broadband method
    - Transient rotor airload method
  - Later
    - High speed thickness noise
    - Non-compact source rotational noise

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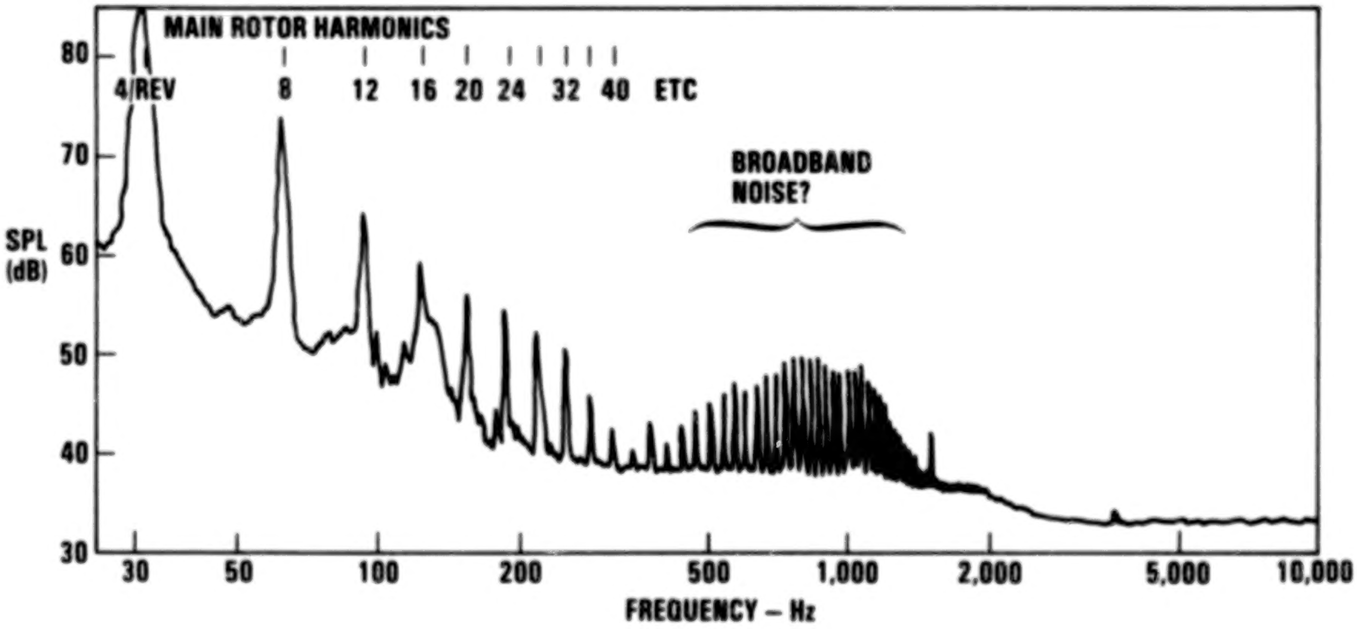
MAIN ROTOR NONROTATIONAL, NONIMPULSIVE NOISE

It has been shown that the so-called main rotor vortex or broadband noise is of primary importance in controlling the annoyance of many helicopter models. Both of these terms are misnomers because it is generally agreed that vortex shedding is not the cause of the subject noise nor is it necessarily broadband as shown in the figure. It is important to understand the nature of this type of noise in order to define the best method of approach to prediction and control.



BROADBAND NOISE SOURCES

A fairly complete list of the possible sources of main rotor broadband noise is shown in the figure. Some have a higher probability of being important than others and others are mutually exclusive. However, it is probable that several of these sources are active on a rotor in any given flight regime. At least partial correlation has been shown for some of the sources in hover, but none have been demonstrated to be effective in cruise flight because of the lack of measured data. It is important that a high quality data base for both hover and cruise flight be generated, that the various theoretical prediction methods be checked against that data, and that a semi-empirical prediction method accounting for the relevant design variables of modern helicopters be formulated and validated.

- **TURBULENCE INGESTION**
  - **LOW FREQUENCY NARROW BAND RANDOM**
  - **HIGH FREQUENCY BROADBAND**
- **SURFACE PRESSURE FLUCTUATIONS**
- **BOUNDARY LAYER INTERACTION WITH THE TRAILING EDGE**
- **STALLED FLOW INTERACTION WITH THE TRAILING EDGE**
- **TIP FLOW INTERACTION WITH THE TRAILING EDGE**
- **LAMINAR FLOW VORTEX SHEDDING**
- **DIRECT RADIATION FROM THE BLADE BOUNDARY LAYER**

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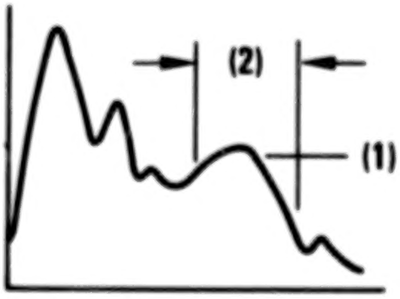
SEMI-EMPIRICAL BROADBAND NOISE PREDICTION

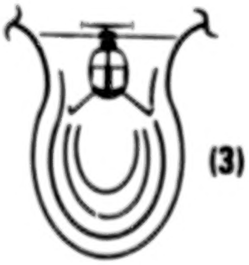
There are three basic requirements for development of a useful semi-empirical broadband noise prediction method. The first is that a methodology, or theoretical framework, be established to account for the parameters which might possibly influence noise generation. The second is that a comprehensive data base be defined. The third requirement is the definition of valid methods for acquiring the necessary data base.

The broadband noise prediction method must account for the parameters peculiar to each particular type of helicopter operation. It may account for only radial aerodynamic variation for conditions such as OGE hover in low-turbulence conditions and will have to use detailed azimuthal variations for the non-hover conditions and hover where there is interference or substantial atmospheric turbulence. All applicable aerodynamic parameters (flow field, airfoil section, tip) must be taken into account as they would in a purely theoretical method, but at a lower degree of computational complexity.

The figure below indicates what should be included in the data base in order to develop and verify such a prediction method. This list is not all inclusive since it includes only one helicopter type and main rotor design. However, it would form a good base on which to build. Acquisition of such a data set is a challenge since it does not exist to date.

- **NEEDED FOR CORRELATION OF WIDE OPERATIONAL SPECTRUM BROADBAND NOISE MODEL**
  - **— MAGNITUDE (1)**
  - **— SPECTRAL DISTRIBUTION (2)**
  - **— DIRECTIVITY (3)**





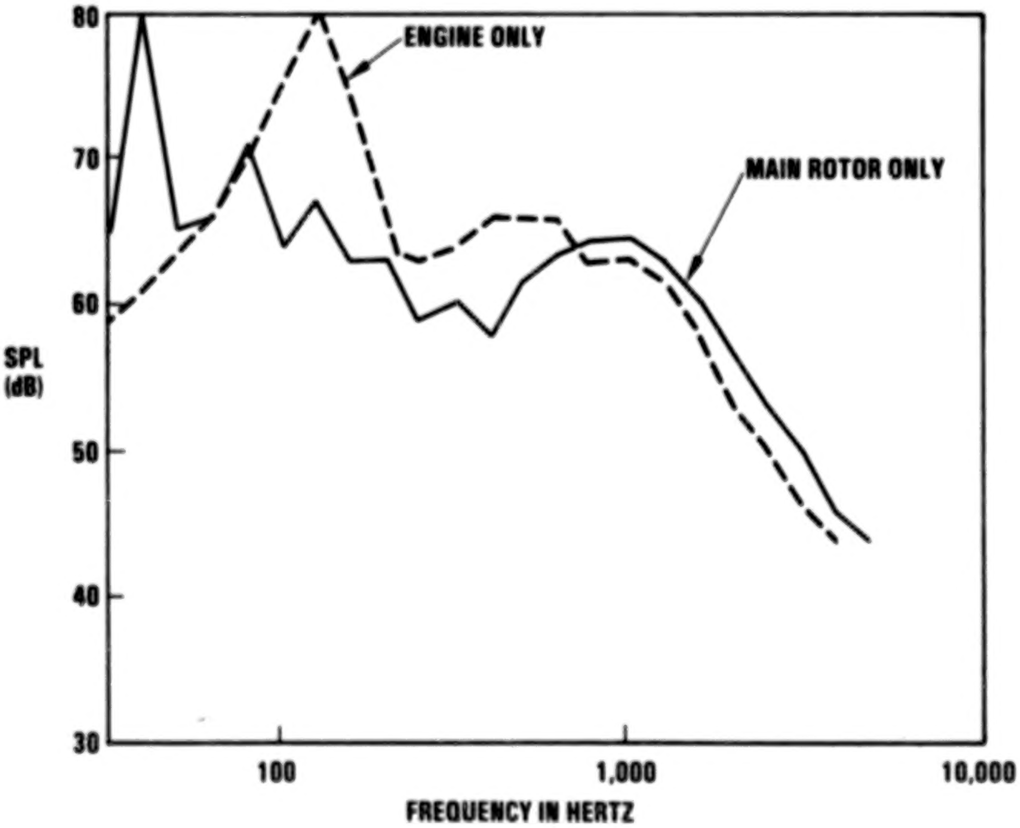
- **OPERATIONAL RANGES FOR WHICH INFORMATION NEEDED**
  - **— HOVER – THRUST/TIP SPEED MATRIX (CORRELATION WITH EXISTING METHODS)**
  - **— CRUISE – 40-120 KT**
  - **— DESCENT – RANGE OF SPEED/ROD**
  - **— CLIMB – RANGE OF SPEED/ROC**

ACQUISITION OF A BROADBAND NOISE DATA BASE

The two methods for acquiring forward flight data on aircraft are the wind tunnel and actual flight testing. The wind tunnel offers the advantages of good control over flight parameters and the opportunity to record many variables in a steady state environment. However, the wind tunnel does not appear very promising for rotor broadband noise at this time for several reasons. The lack of scaling relationships precludes the use of small scale data from current anechoic free-jet type tunnels. Full scale tunnels do not have adequate acoustic treatment to produce the free-field environment simulation required for broadband noise measurement. The signal to noise ratios are generally inadequate due to this lack of treated facilities and the broadband nature of the noise makes frequency discrimination impossible. Near field measurements (useful in increasing signal to noise ratio) are not practical because of the distributed source characteristics.

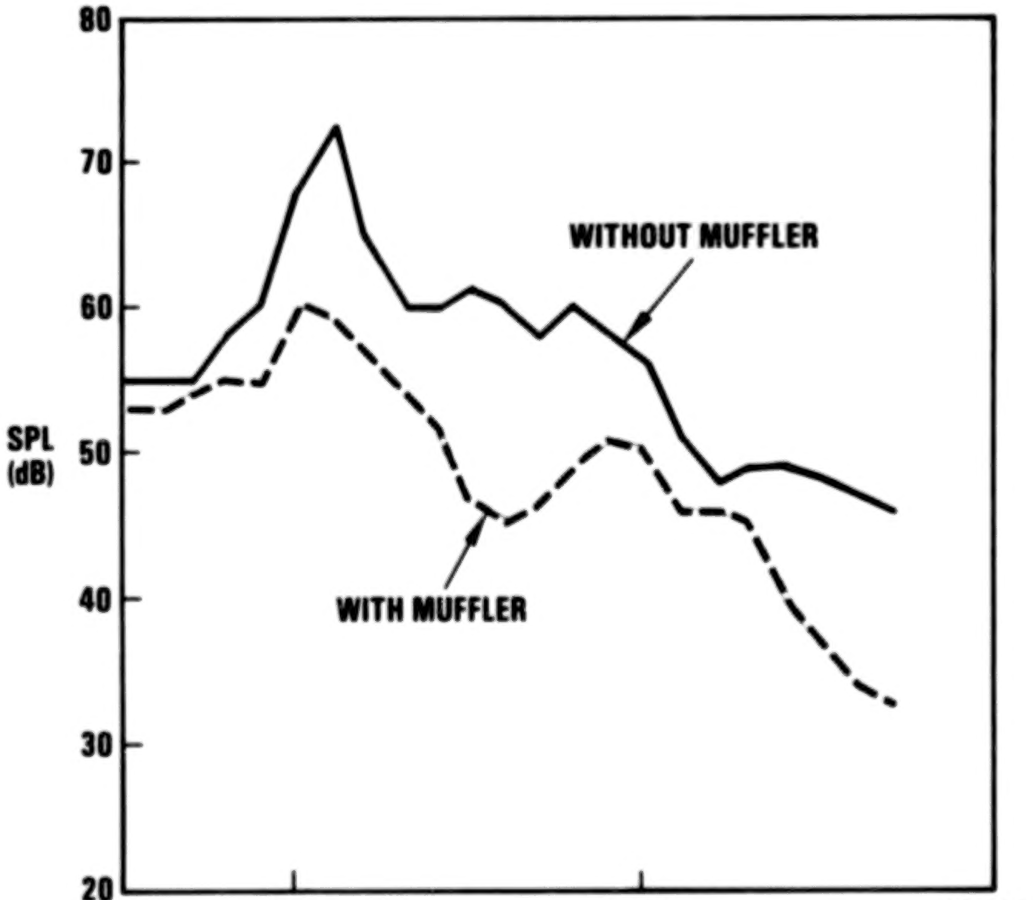
Flight test for main rotor broadband noise measurement has its own shortcomings. These include interfering noise generated by the engine and tail rotor, and the necessity of ground microphone measurements which make the rotor a transient rather than steady state noise source. However, the former problem can be overcome with suppression of the conflicting noise sources and the latter problem can be either turned into an advantage (obtaining directivity data) or avoided by using air to air data acquisition.

The main problem in flight test main rotor broadband noise measurement is in the masking noise generation of other sources. The figure below shows that the engine and main rotor generate similar mid-frequency noise signatures for an OH-6A helicopter.



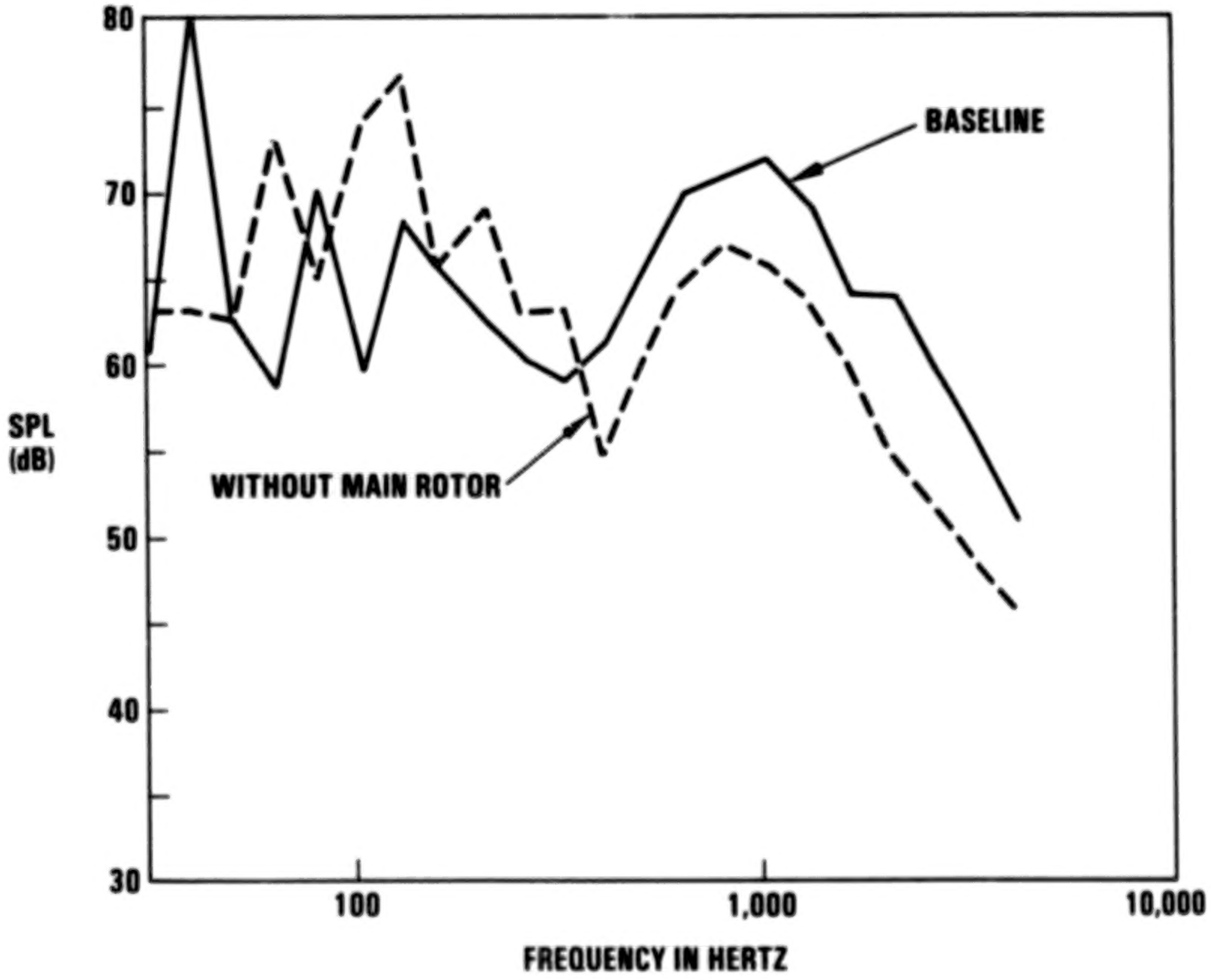
TURBINE ENGINE SILENCING

Silencing of helicopter mounted turboshaft engines has been accomplished only rarely in the past. The reasons include lack of need, weight restrictions, space restrictions and the lower frequency character of the noise. Hughes Helicopters applied both dissipative and reactive muffling to an OH-6A helicopter during the quiet helicopter program in the early 1970s to attain the attenuation shown in the figure. Attenuation was biased toward the lower frequency end of the spectrum by design for this study since it was aimed at aural detection rather than reduced annoyance. Improvement of the design for higher attenuation in the 500 to 2000 Hz range would be easily accomplished to provide much lower masking levels for main rotor noise evaluation.



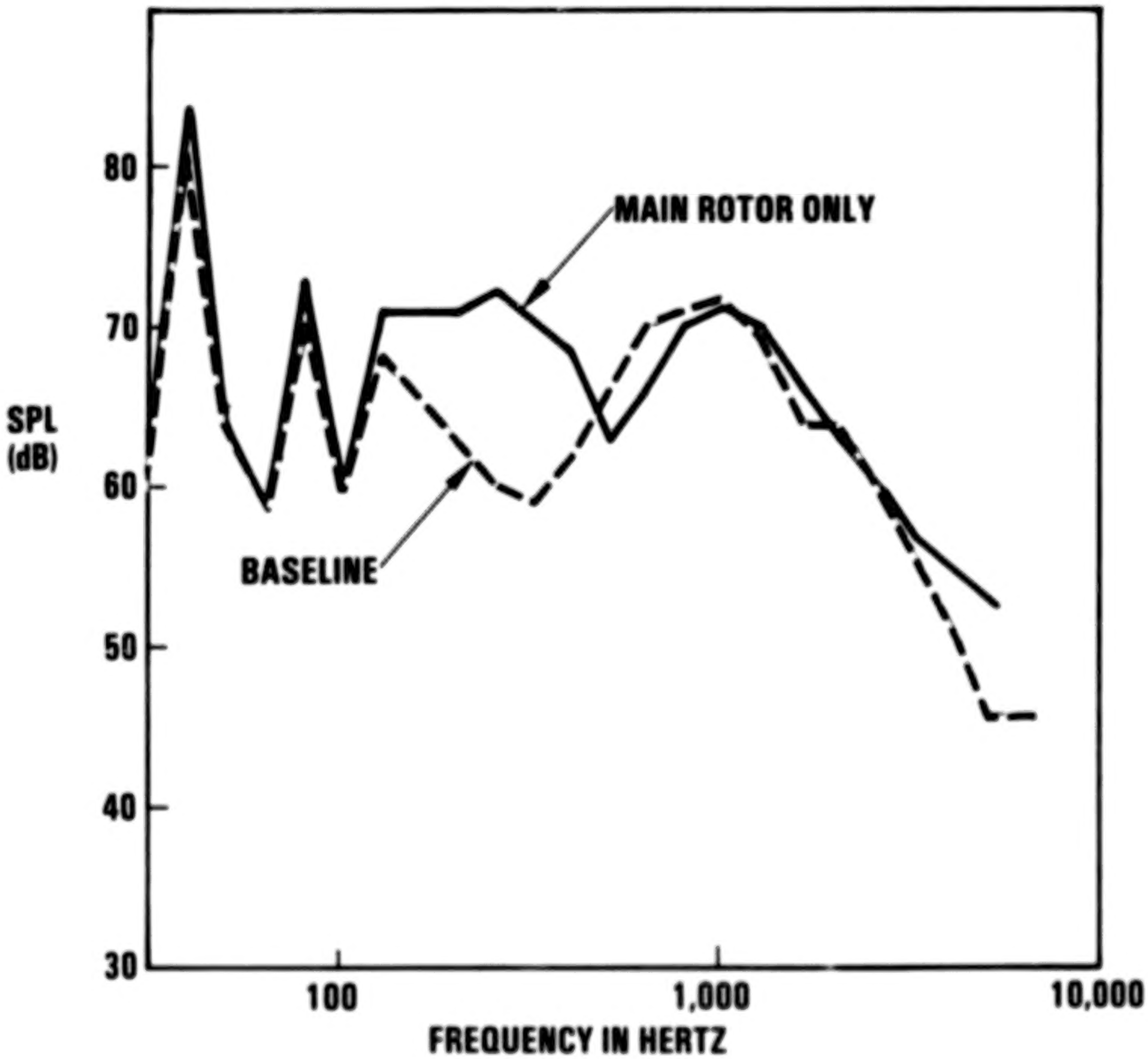
DOMINANCE OF MAIN ROTOR BROADBAND NOISE WITHOUT ENGINE

The figure shows OH-6A noise for the combined main-tail rotor in operation (labeled Baseline) and for tail rotor operation only. The engine was silenced with a large tank silencer connected through an insulated duct in each case. It appears that, even for the standard tail rotor installation, the main rotor is dominant in the 500 to 2000 Hz frequency range of interest and that valid main rotor broadband noise data could be acquired. This is especially true since the microphone location at which this data was taken favors tail rotor rotational noise. It was located approximately eight feet below the rotors and thirty degrees left of the aft centerline of the helicopter.



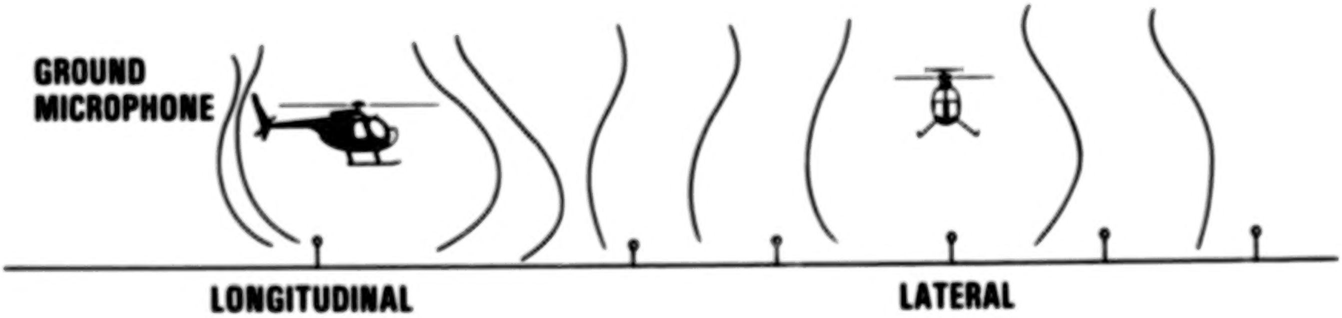
THE MAIN ROTOR IS NEARLY DOMINANT FOR THE BASELINE CASE

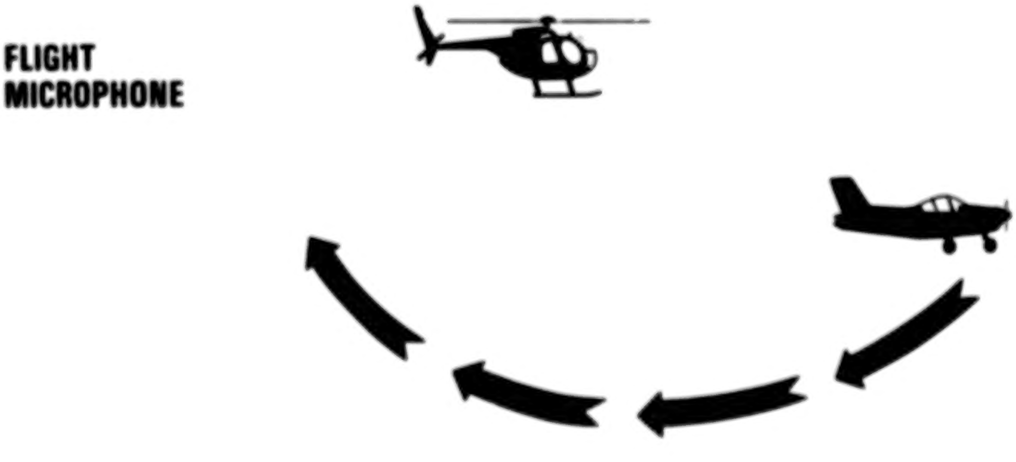
Even with the engine unmuffled and the tail rotor operating at normal thrust, the main rotor is nearly dominant for the broadband noise frequency range as shown in the figure. The fact that the main rotor alone generates more noise than the total baseline helicopter in the 100 to 500 Hz frequency range in the figure is an interesting point which is not felt to be generally significant. What it does indicate is that rotor noise, particularly higher harmonic content, is quite variable with subtle operating condition changes making the prediction task very difficult.



MAIN ROTOR BROADBAND NOISE MEASUREMENT

Once the engine and tail rotor noise has been suppressed adequately to provide main rotor broadband noise dominance for the proper frequency range and directivities, the flight testing becomes rather straightforward. Spectrum characteristics as a function of directivity can be determined via flights over a microphone array as shown in the upper portion of the figure in various flight conditions. Rotor geometry and operating conditions should be varied also if possible to form a more generally useful data base. It is desirable for some flight conditions to acquire data in the steady state for more detailed analysis than is possible with flyover information. The lower portion of the figure suggests that this data can be acquired with the NASA YO-3 microphone system using the new station-keeping system for positioning. The new system will allow data acquisition at a number of relative (helicopter to microphone system) locations that have not been practical in the past and which are particularly important to this type of noise.





SUMMARY

There are several important points to be emphasized in this presentation and they are listed in the figure below. It is felt that main rotor noise is the key to quieter helicopters since there are proven and relatively inexpensive ways to handle the tail rotor. Blade Vortex Interaction (BVI) noise is a problem to some extent with all helicopters, particularly in the descent mode. Methodology must be developed to allow forecast and control of this type of noise. Additional means of controlling it, such as reduced rotor speeds for terminal operations (two-speed transmission of wide range rotor speed operation) should also be pursued because they may be the most effective means of control and they apply to all helicopter models. Main rotor broadband noise is the limiting factor in overall helicopter noise generation for many models now and will be more so in the future as tail rotor noise control is applied by the industry. Development of semi-empirical methods to predict its behavior is necessary if results are to be achieved in the short time period available. These methods cannot be developed (nor can the first-principles methods be sorted and validated) without an extensive data base. This urgently needed data base does not exist now, so high priority must be assigned to its acquisition. Flight test is the only way that is currently practical for achieving this objective and methodology has been suggested for acquiring good quality data.

- **CONTROL OF MAIN ROTOR NOISE IS CRITICAL**
  - **BVI – APPROACH**
  - **BROADBAND – OTHERS**

- **SEMI-EMPIRICAL METHODS ARE NECESSARY INTERIM MEANS OF CONTROL**

- **NO NON-HOVER DATA BASE EXISTS**

- **DATA BASE POSSIBLE VIA FLT TEST**

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FACILITY REQUIREMENTS FOR HELICOPTER NOISE RESEARCH

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ABSTRACT

Development of future helicopter noise prediction methods requires an accurate experimental data base for correlation studies or assessment of theoretical prediction methods. This presentation defines the aerodynamic and acoustic performance criteria for conducting experimental studies of helicopter noise using existing facilities. Requirements for ground-based facilities are described in addition to the limitations associated with full-scale and model-scale studies. Flight testing methods are also evaluated briefly and the restrictions associated with these approaches are cited. Finally, a general evaluation of ground-based and flight testing methods is given. Based on this presentation, future investigators can select the experimental approach best suited for generating a desired data base. Also, facility improvements needed to extend the state of the art in helicopter noise experimental studies can be identified.

OBJECTIVES OF EXPERIMENTAL STUDY

Performance criteria for experimental facilities are a function of the objectives of the study to be conducted. These objectives range from certification of full-scale helicopters to providing local aerodynamic measurements for input to aeroacoustic theories. The experimental facilities needed to achieve these objectives vary from full-scale flyover tests, in the case of certification measurements, to isolated rotor or isolated airfoil studies. These vastly different approaches reflect the difference between product assessment and basic research efforts. But most important, these differences control the individual facility performance requirements.

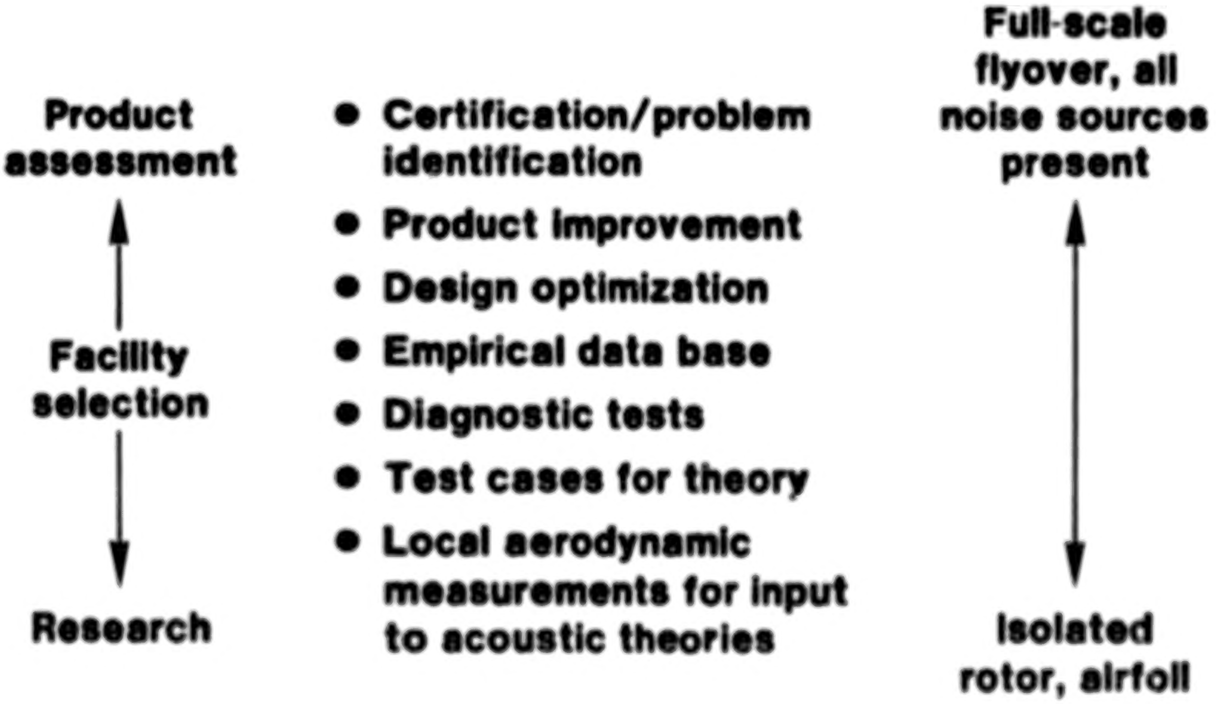
Product assessment
Facility selection
Research
Certification/problem identification
Product improvement
Design optimization
Empirical data base
Diagnostic tests
Test cases for theory
Local aerodynamic measurements for input to acoustic theories
Full-scale flyover, all noise sources present
Isolated rotor, airfoil

Figure 1

ACOUSTIC SOURCES AND AVAILABLE TEST FACILITIES

Facility performance criteria are also controlled by the particular acoustic source which is to be investigated. The sources range from steady loading noise to unsteady periodic forces. In each case the measured acoustic field depends on the rotor operating conditions, blade geometry and source directivity. These sources can be investigated using a ground-based wind tunnel, open-jet acoustic wind tunnel, or hover stand. Full-scale or model-scale rotors can be employed in these different facilities. Finally, flyover and in-flight testing methods can be used.

- **Blade force noise**
  - **Rotating steady forces (Gutin)**
  - **Unsteady periodic forces (BVI, stall, shock)**
  - **Unsteady random forces (turb. ingestion, blade self noise)**
- **Blade volume noise**
- **Quadrupole radiation**

In each case sound sensed by observer depends on:

Rotor operating conditions
Blade geometry
Directivity

Figure 2

CRITERIA FOR SELECTING FACILITIES

Criteria for selecting the appropriate experimental facility include the objectives of the study (product assessment or research), the acoustic source type, and the numerous rotor performance parameters which are to be investigated. The last category includes geometric, aeroelastic and performance scaling effects. For example, rotor or isolated airfoil studies designed to investigate trailing-edge noise must simulate the full-scale Reynolds number since this source mechanism is controlled by the viscous boundary-layer characteristics. Finally, the facility schedule and facility characteristics influence the selection process.

The remainder of this paper describes the facility performance characteristics required to obtain an accurate helicopter noise data base. Aerodynamic and acoustic performance criteria of wind tunnels, open-jet acoustic tunnels and acoustically treated hover stands are described first using specific examples. Flyover and in-flight test facilities are discussed later.

- **Needs/objectives**
  - **Research**
  - **Certification**
  - **Piggyback with minimum costs**
- **Acoustic source type**
- **Rotor geometric scaling effects**
  - **AR, t/c, T, N, profile shape**
- **Rotor performance scaling effects**
  - **Re, M_t, ADVR, C_T/σ**
- **Blade aeroelastic scaling effects**
- **Instrumented blades**
- **Flow field measurements**
- **Cost versus data repeatability**
- **Facility availability**
- **Facility requirements/characteristics**

Figure 3

GENERAL AERODYNAMIC PERFORMANCE REQUIREMENTS OF WIND TUNNELS, OPEN-JET ACOUSTIC TUNNELS, AND HOVER STANDS

Several aerodynamic performance requirements should be satisfied to ensure an accurate helicopter noise data base. For example, a 1/2-percent variation in mean velocity results in approximately a 1-percent change in the freestream dynamic pressure experienced by the rotor. To minimize fluctuations in the rotor operating conditions, the mean velocity should be uniform and steady. Mean flow deflections should also be minimized to avoid changing the rotor angle of attack and freestream dynamic pressure. In addition, the freestream turbulence intensity should be low to avoid generating extraneous turbulence ingestion noise. Finally, when necessary, the test vehicle operating parameters should be accurately sensed and controlled to ensure realistic simulation of helicopter flight conditions. Specific examples of these aerodynamic requirements are presented in the following figures.

- **Flow uniformity (within 1/2%)**
- **Mean flow unsteadiness (0.5%, <0.2% pref)**
- **Turbulence level (0.5%, <0.2% pref)**
- **Wind-tunnel wall effects/open jet deflection minimal**
- **Sense and control rotor operating conditions accurately**

Figure 4

THE EFFECT OF WIND-TUNNEL WALLS ON ROTOR AERODYNAMIC OPERATING CONDITIONS

Mean flow deflections due to the presence of a lifting rotor in a wind tunnel have a measurable impact on the blade angle of attack (α) and inflow dynamic pressure (q). This is illustrated by the incremental changes in α and q for a rotor operating in a closed and open (floor in place) wind tunnel. The reference condition for these calculations is a rotor in free flight. The results demonstrate that the desired operating condition may not agree with the assumed free-flight condition. Thus, the influence of wall effects is significant and must be accounted for analytically when specifying the rotor aerodynamic operating conditions.

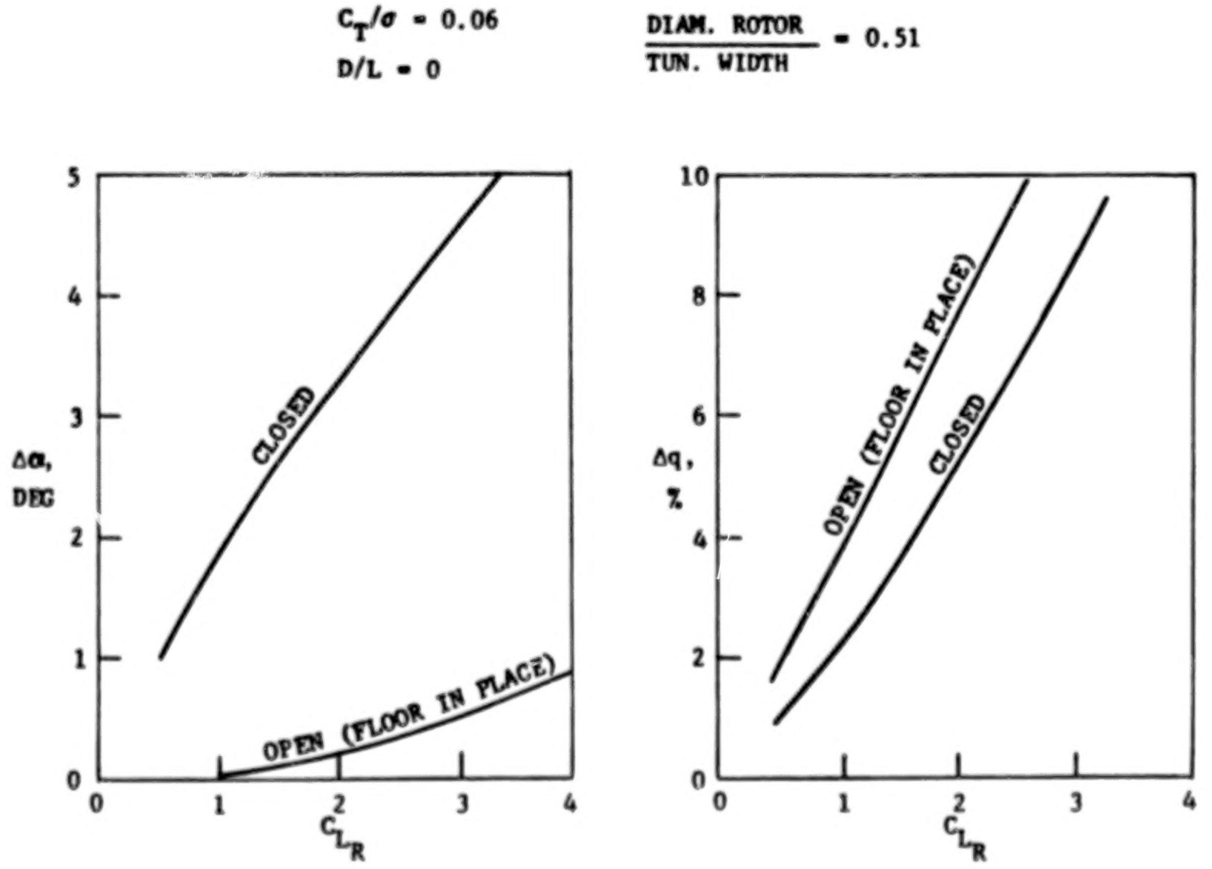

Figure 5

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THE EFFECT OF OPEN-JET DEFLECTION ON ROTOR OPERATING CONDITIONS AND INGESTION OF SHEAR LAYER

Mean flow deflections can also occur in an open-jet acoustic wind tunnel when a rotor is operated at high thrust conditions. Similar to the wind-tunnel case, the rotor aerodynamic operating conditions are changed. Another potentially detrimental effect is the ingestion of the open jet-shear layer. This results in the generation of turbulence ingestion noise (described below) and should be avoided. A hot wire can be used to check the unsteady characteristics of the velocity field immediately upstream of the rotor plane. Mean velocity profile measurements are inadequate for determining the edge of the intermittent shear layer.

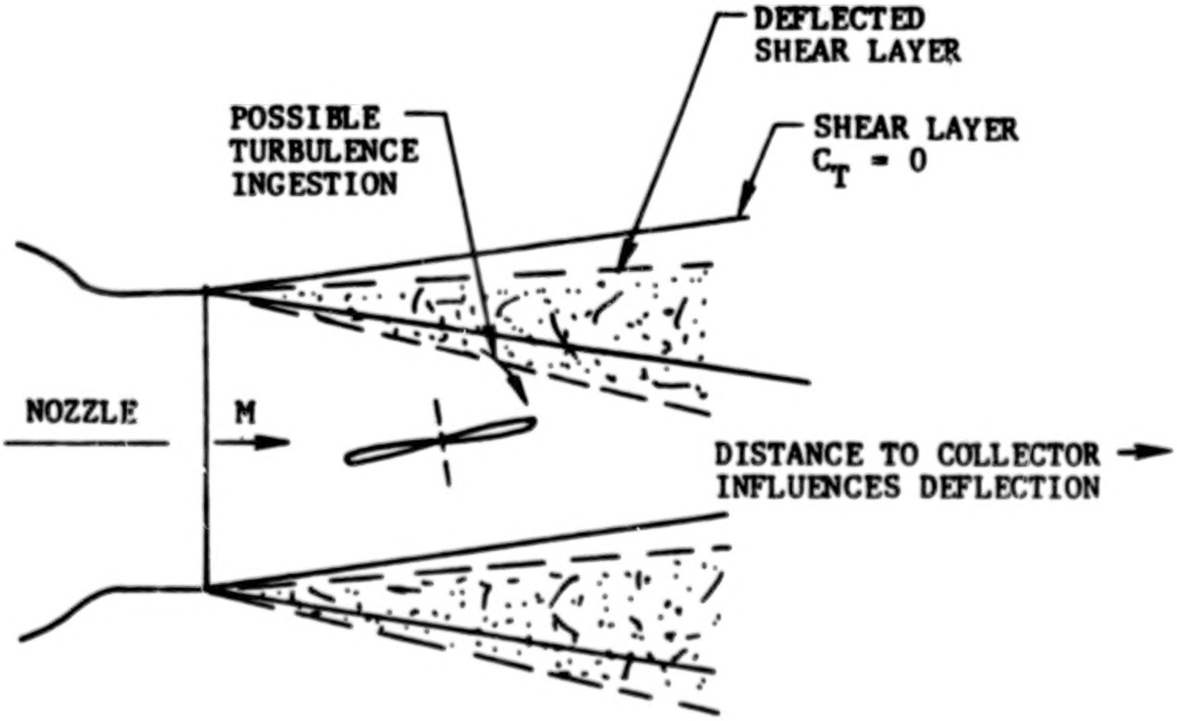
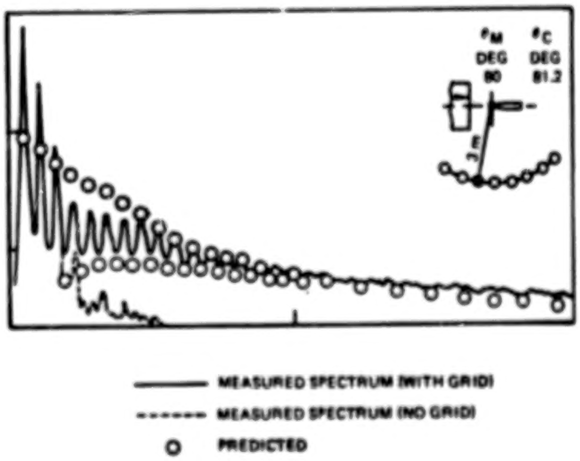

DEFLECTED SHEAR LAYER
POSSIBLE TURBULENCE INGESTION
SHEAR LAYER $C_T = 0$
NOZZLE
M
DISTANCE TO COLLECTOR INFLUENCES DEFLECTION

Figure 6

TURBULENCE INGESTION NOISE

Ingestion of turbulence results in the generation of blade passing harmonics at low frequencies and broadband noise at high frequencies. This is illustrated in the spectrum shown here for an isolated rotor operating in a grid-generated turbulence field (Ref. 1). For comparison, a second measurement shows the noise generated in the absence of the grid. Such spurious noise contributions, generated by various facility turbulence sources, should be minimized. The parameters which must be controlled are the transverse (normal to the chord) turbulence component and the axial length scale of the turbulence field (Ref. 1).



TURBULENCE SOURCES

RECIRCULATION OF WAKES (CLOSED LOOP)

SHARP CORNERS (CLOSED LOOP)

BOUNDARY LAYERS

SEPARATED FLOW

SUPPORT STRUT WAKES

↓ NEED

TURBULENCE SUPPRESSION SCREENS

STRONG CONTRACTION

Figure 7

SENSING AND CONTROLLING ROTOR OPERATING CONDITIONS

Helicopter operating conditions, such as thrust and drag, are often important parameters in a test program. In this case, accurate sensing and control of the rotor parameters is required to ensure realistic simulation of flight conditions. The required accuracy of such measurements is illustrated with the example presented here.

EXAMPLE FOR BLADE VORTEX INTERACTION (BVI)

GIVEN: UH-IH, OPERATING 80 KNOTS
BVI OCCURS FOR $300 < V_Y < 580$ ft/min

CONSIDER: 12 FT DIA. MODEL ROTOR IN VSTOL TUNNEL,
DESCENT ANGLE VARIATION $\Delta\gamma = 1.98^{\circ}$
PROPULSIVE FORCE VARIATION $\Delta C_D = 20.7$ lbs

TO RESOLVE 10 POINTS OVER $\Delta\gamma$, MUST RESOLVE AND CONTROL
$\Delta C_D = 2.1$ lbs DRAG IN 485 lbs THRUST

Figure 8

GENERAL ACOUSTIC PERFORMANCE REQUIREMENTS OF WIND TUNNELS, OPEN-JET ACOUSTIC TUNNELS, AND HOVER STANDS

Numerous acoustic performance requirements should also be satisfied to ensure an accurate rotor noise data base. A list of the requirements is presented here with specific examples given in the following figures.

- **MICROPHONE LOCATED IN ACOUSTIC AND GEOMETRIC FAR FIELD**
- **IN-FLOW MICROPHONE SELF NOISE <BACKGROUND NOISE**
- **IN-FLOW MICROPHONE DIRECTIONAL SENSITIVITY ACCOUNTED FOR (CAN BE CALIBRATED)**
- **EFFECTS OF PROPAGATION THROUGH SHEAR LAYER INCLUDED IN DATA REDUCTION**
- **FIELD OF VIEW ADEQUATE FOR SOURCE TYPE**
- **ACOUSTIC LINING CUT-OFF FREQUENCY <SOURCE FREQUENCY (SIGNIFICANT LIMITATION IN SOME FACILITIES)**
- **UNIFORM FREE FIELD DECAY IN MULTIPLE DIRECTIONS (VARIATION <±1dB)**
- **BACKROUND NOISE 10 dB BELOW SOURCE NOISE (OTHER UNCERTAINTIES CAN REDUCE MARGIN)**
- **CONVERSION OF DATA TO FLYOVER CONDITIONS (REQUIRES MOVING SOURCE WITH GROUND REFLECTION)**

Figure 9

MICROPHONE LOCATION

Microphones used to measure the noise generated by a rotor should be located in the geometric and acoustic far field. Satisfying this requirement depends on the relative dimensions of the facility in addition to the source frequency. Open-jet facilities typically satisfy the geometric far-field condition because of the small rotor size compared to the dimensions of the surrounding anechoic chamber. Wind tunnels are marginal in meeting this requirement since the test rotors are usually large. A consequence of this situation is illustrated here for a microphone distance typically used by investigators. The apparent microphone angular position relative to the rotor hub corresponds to 45° while the true directivity pattern sensed by the microphone corresponds to 62°. This demonstrates that erroneous source directivity characteristics can be measured in the geometric near-field unless the source position is known. The error increases for distributed sources. In this case, experimental results can only be compared with geometric near field predictions of the rotor noise. Extrapolations to the acoustic far field must be obtained from analytical predictions.

The acoustic far-field requirement can be determined from the characteristics of a dipole. The far field is established when the measurement distance is larger than $\lambda/6$. However, a distance of one wavelength is preferred.

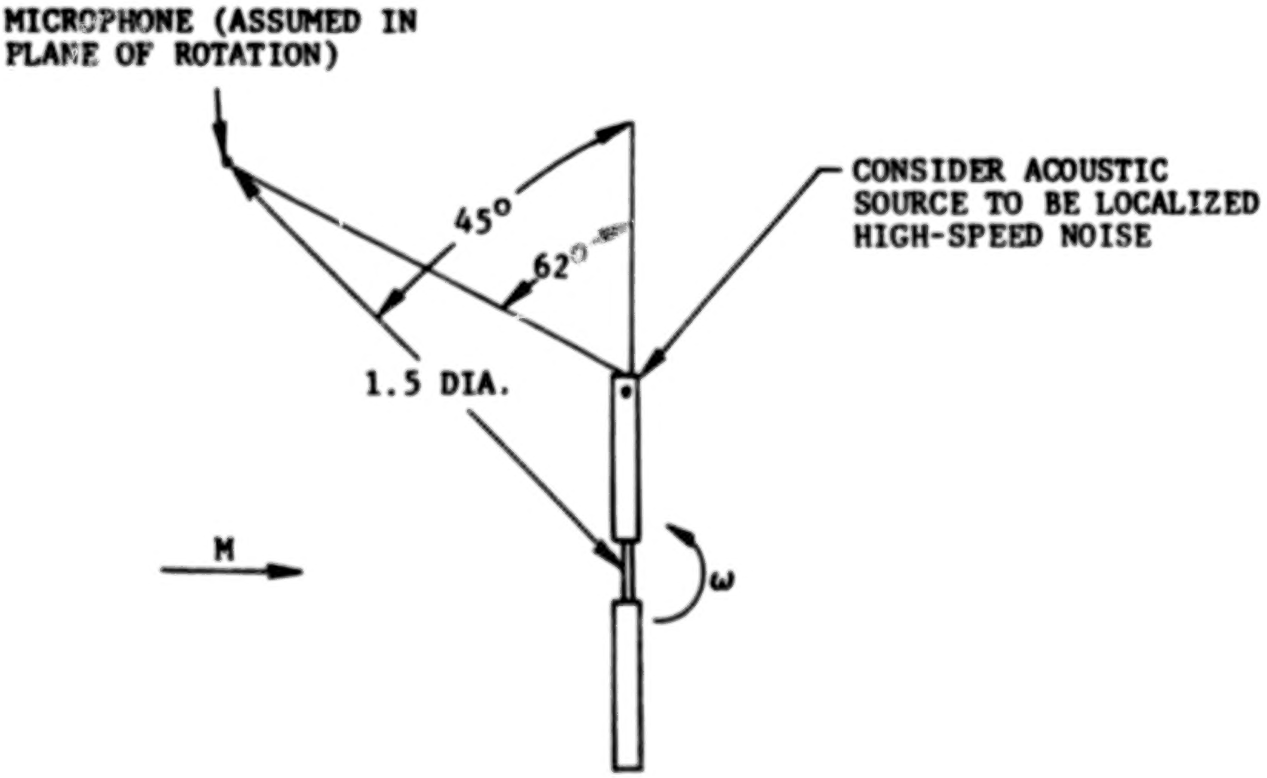

MICROPHONE (ASSUMED IN PLANE OF ROTATION)
45°
62°
CONSIDER ACOUSTIC SOURCE TO BE LOCALIZED HIGH-SPEED NOISE
1.5 DIA.
M
ω

Figure 10

IN-FLOW MICROPHONE SELF NOISE

Helicopter acoustic studies conducted in wind tunnels often require placing microphones in the flow field. The pressure fluctuations sensed by the microphone contain contributions from the acoustic source and self-noise sources which introduce broadband or discrete tone (vibration of support stand) characteristics in the measured spectrum. It is important to recognize that the self noise could control the microphone signal-to-noise ratio. In this case, the use of acoustic treatment to reduce the facility background noise has no impact. The parameters controlling the various self-noise mechanisms are unknown and can only be postulated. Thus, scaling data to obtain self-noise estimates at different velocities represents, at best, an estimate.

SOURCES OF SELF NOISE

- **TURBULENCE IN FLOW**
- **TURBULENT BOUNDARY LAYER ON NOSE CONE**
- **SEPARATED FLOW ON NOSE CONE AT NON-ZERO INCIDENCE ANGLES**
- **NOISE RADIATED BY SUPPORT STAND**
- **VIBRATION OF SUPPORT STAND**

Figure 11

TYPICAL EXAMPLE OF MICROPHONE SELF NOISE

A spectrum calibration curve for the self noise of a microphone with a nose cone demonstrates the magnitude of the problem. Since the spectrum amplitude at the highest velocity is comparable to the sound pressure level reported in some helicopter wind-tunnel studies, it is important that self-noise sources be minimized in future experimental studies. One approach employs a specially designed turbulence suppression screen which replaces the standard nose cone. The frequency response calibration curve shows a dramatically reduced sensitivity to self noise. However, this advantage is offset by the non-uniform directional sensitivity which must now be accounted for when placing microphones in the forward measuring arc, as described in Figure 13.

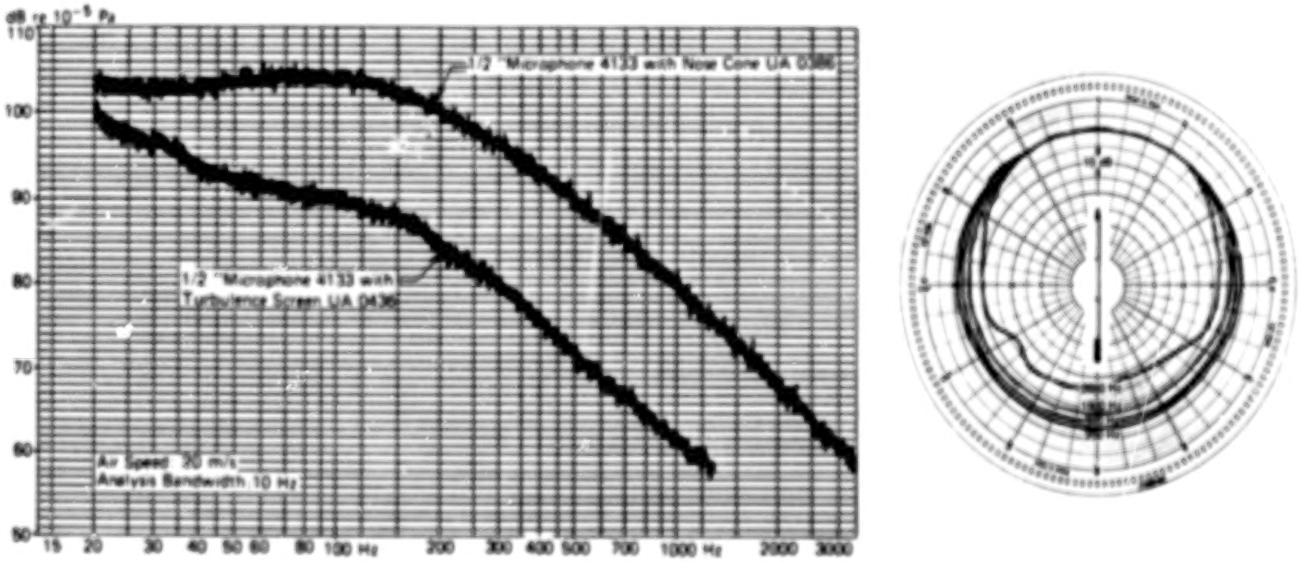


Figure 12

IN-FLOW MICROPHONE DIRECTIONAL SENSITIVITY

Wind-tunnel acoustic measurements obtained in the foreward arc of a helicopter model require locating the microphone upstream of the noise source region. Acoustic wavefronts now arrive at oblique angles relative to the microphone centerline. The resulting measurement can be influenced by the nose-cone directional sensitivity in addition to diffraction effects at high frequencies. These features must be accounted for using microphone calibrations to ensure accurate rotor noise directivity measurements.

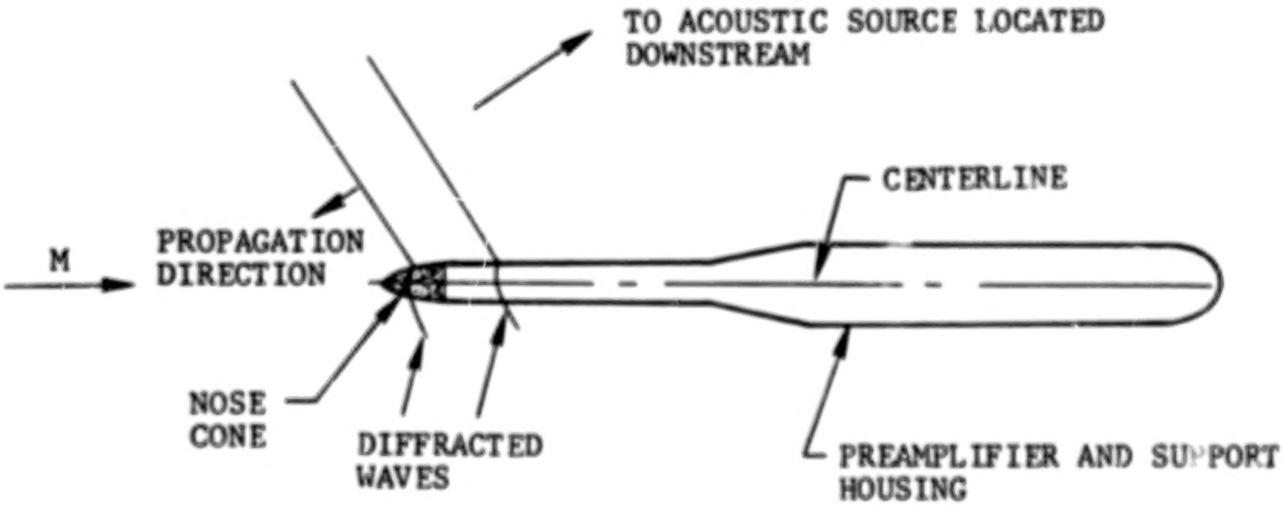

TO ACOUSTIC SOURCE LOCATED DOWNSTREAM
CENTERLINE
M
PROPAGATION DIRECTION
NOSE CONE
DIFFRACTED WAVES
PREAMPLIFIER AND SUPPORT HOUSING

Figure 13

MEASURED NOSE-CONE DIRECTIONAL SENSITIVITY

The importance of the microphone directional sensitivity is demonstrated by the calibration curves for a microphone fitted with a nose cone. While the directional sensitivity is weak below 5 KHz, the magnitude of the microphone correction is significant at higher frequencies. Note that the microphone orientation is defined by the angle between the centerline in Figure 13 and a vector in the direction of the acoustic source.

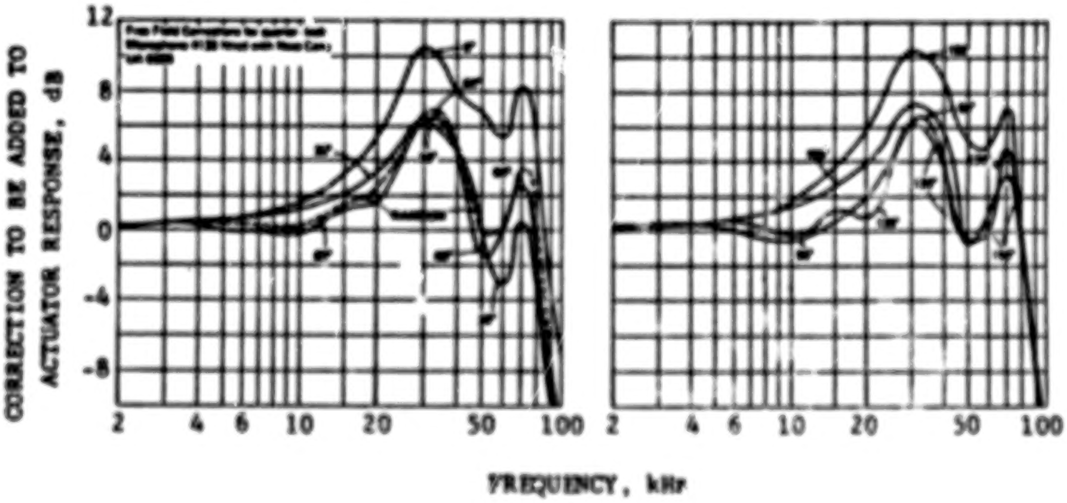
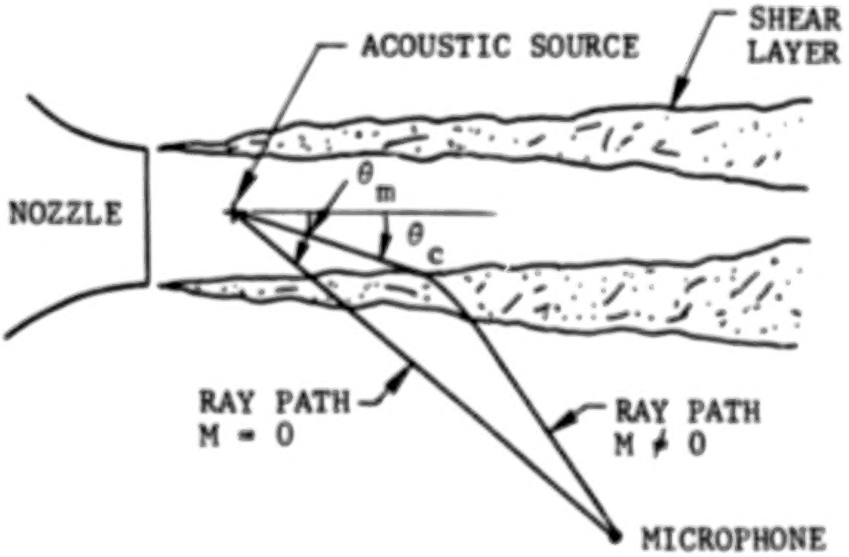


Figure 14

SOUND PROPAGATION THROUGH SHEAR LAYERS

Acoustic studies performed in open-jet wind tunnels result in the transmission of sound through the jet shear layer. For tests conducted at free-stream Mach numbers less than 0.1, measurements outside the airstream can be used directly to infer the source noise characteristics. However, at high Mach numbers, the open-jet technique is susceptible to errors due to the shear layer refraction, reflection, and scattering of the sound radiated by the model. These effects significantly alter the conclusions drawn from a particular experiment. This point can be demonstrated for the refraction phenomenon. Acoustic ray paths for the $M = 0$ and $M \neq 0$ jet operating conditions arrive at the same out-of-flow microphone. But, each ray path defines a different radiation direction indicating the discrepancy introduced by refraction. This phenomenon should be accounted for if accurate acoustic data is expected from a test program. Reference 2 provides a verified analytical procedure to correct for refraction effects.



SOUND ARRIVING AT MICROPHONE IS
REFRACTED AND SCATTERED

Figure 15

REFRACTION ANGLE CORRECTION

Refraction angle changes across the shear layer can be predicted as a function of Mach number. Angle θ_c defines the radiation angle on the ray path corresponding to $M \neq 0$ (relative to the downstream axis) while θ_m represents the measurement angle at $M = 0$. These curves, obtained from Reference 2, can be used to estimate when refraction angle corrections are needed during data reduction.

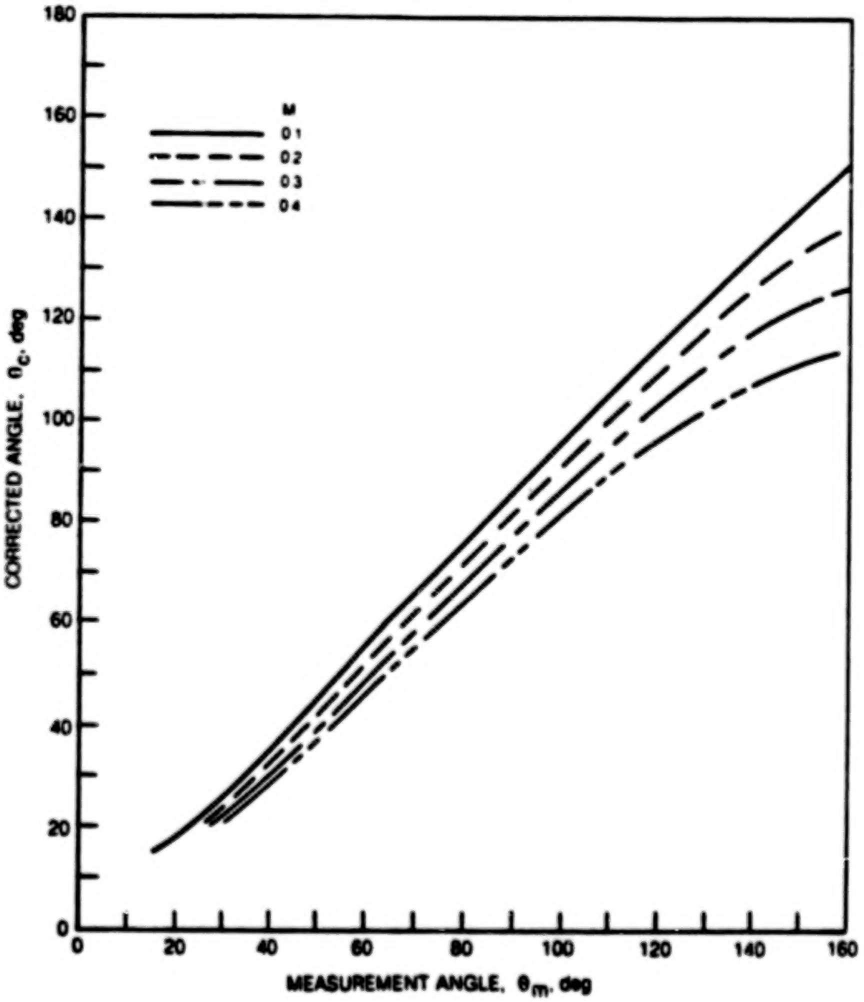


Figure 16

REFRACTION AMPLITUDE CORRECTION

In addition to an angle change, refraction also introduces an amplitude change. This occurs because of the larger propagation distance for the M ≠ 0 case (Fig. 15), in addition to a change in the acoustic wavefront sperical divergence across the shear layer. An example (Ref. 2) of the amplitude correction which must be applied during data reduction is shown in Figure 17. The magnitude of this correction illustrates the importance of the problem.

It should be noted that the refraction effects described in Figures 16 and 17 apply only for sources located on the axis of the open jet. Most helicopter noise sources are distributed over the rotor disk thereby representing off-axis sources. Localized sources related to high-speed noise or blade slap also occur at off-axis locations. This geometric complication, which is discussed in Reference 2, has a strong impact on sound measured in the downstream quadrant.

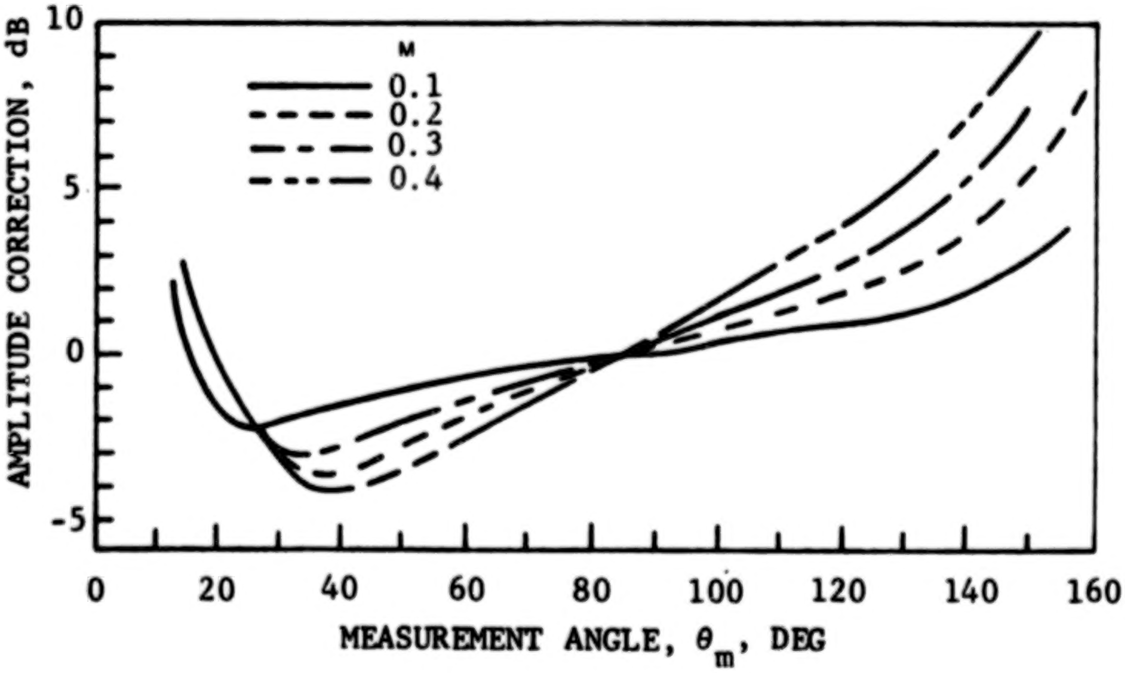


Figure 17

SHEAR LAYER SCATTERING EFFECT

Propagation of rotor discrete tone noise through an open-jet shear layer can create spectral broadening (Ref. 2). This effect is apparent when a discrete tone measured inside the open jet is compared with a measurement outside the flow. Tone broadening occurs for upstream or downstream propagation angles because of a Doppler shift introduced by the flow. In the presence of this phenomenon, peak amplitudes in a narrowband spectrum are decreased since energy is transferred to adjacent bands. Based on the study in Reference 2 this effect can be accounted for by integrating the area under each discrete tone to recover the total acoustic energy. The circles in Figure 18 verify this conclusion for the broadened tones measured at different radiation angles, θ_c.

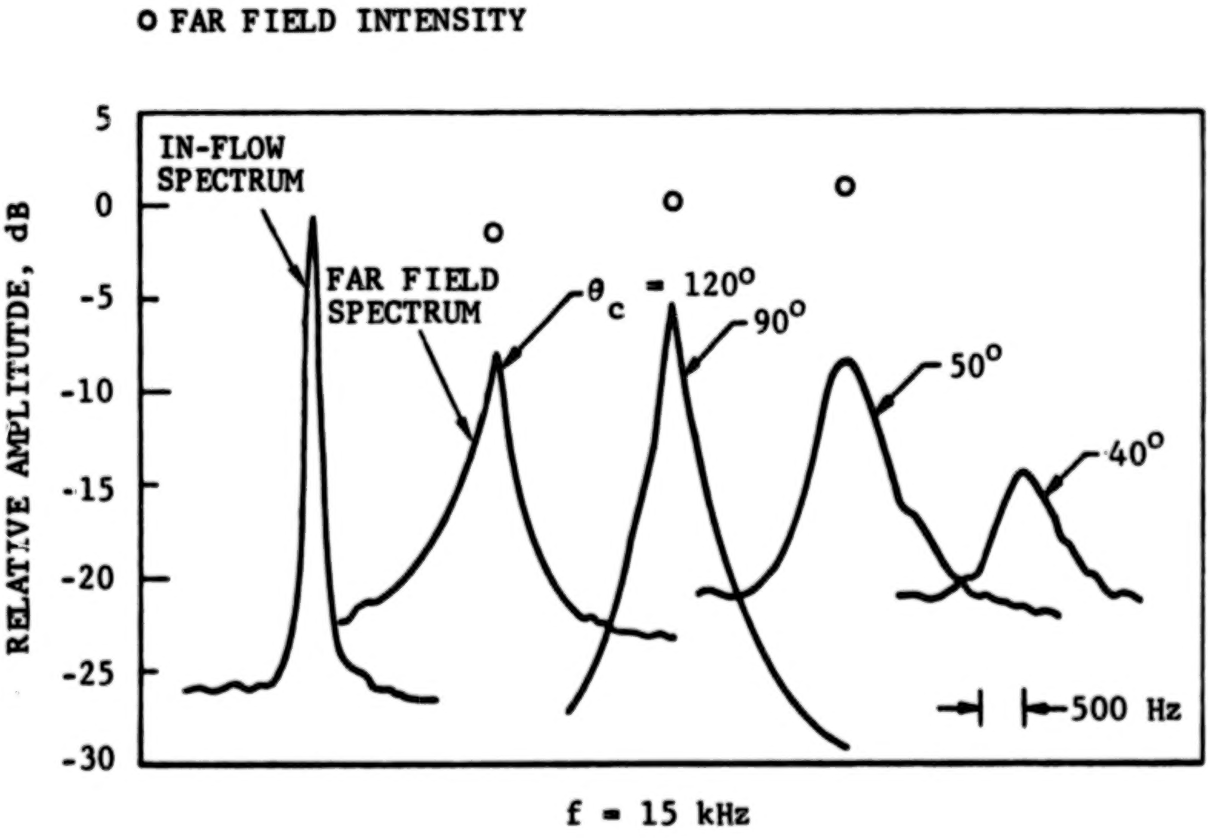


Figure 18

FIELD OF VIEW IN OPEN-JET FACILITIES

The field of view in open-jet wind tunnels is controlled by the location of the upstream nozzle, the downstream collector, and the facility walls. These physical restraints can limit the range of the source directivity measurements as shown by the scaled figure for an existing facility. Thus, it is possible that acoustic sources which radiate predominantly in the forward arc (such as blade slap during descent) cannot be documented completely with microphones outside the flow. In this case, microphones must be placed inside the open jet nozzle. This introduces measurement uncertainties due to potential reflections and reverberation inside the nozzle. Also, the wake from the microphone support stand can be convected downstream into the rotor resulting in extraneous noise generation.

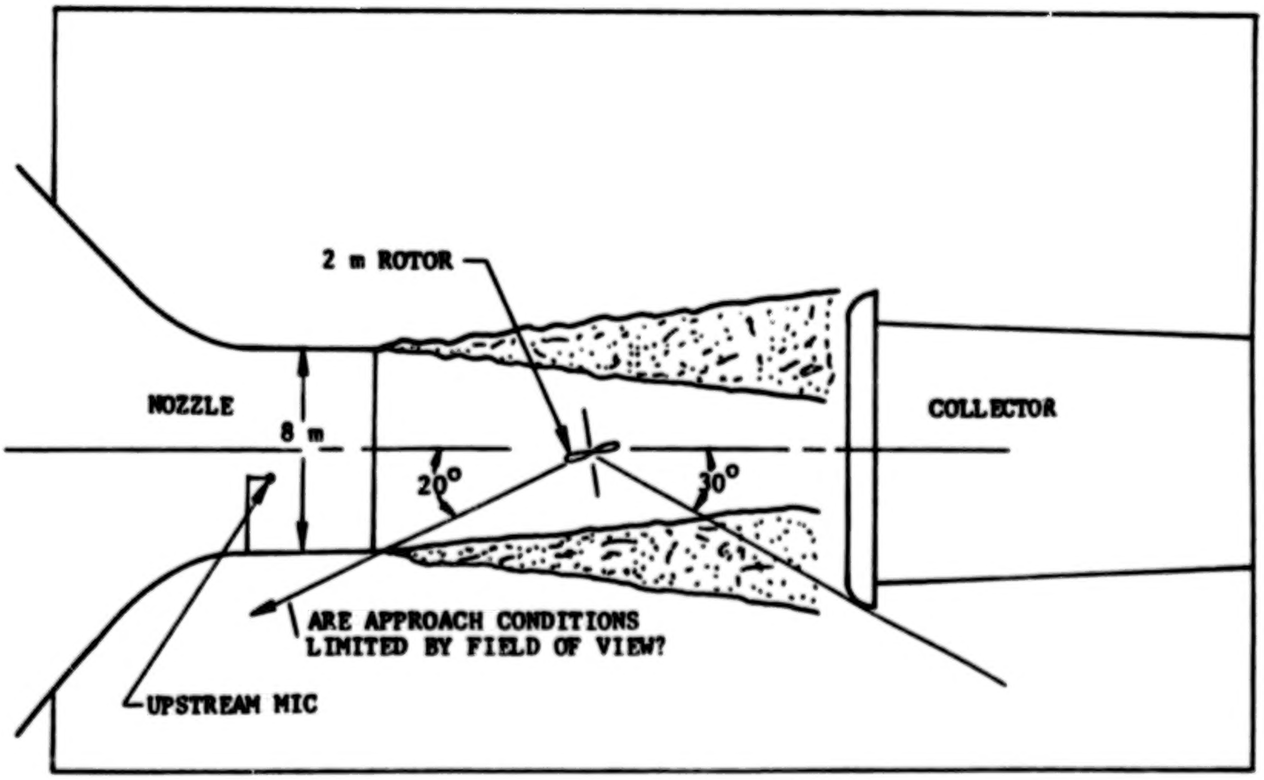

2 m ROTOR
NOZZLE
8 m
20°
30°
COLLECTOR
ARE APPROACH CONDITIONS
LIMITED BY FIELD OF VIEW?
UPSTREAM MIC

Figure 19

FIELD OF VIEW IN WIND TUNNEL

Similar to the open-jet facilities, wind tunnels are limited in their field of view. Measurements on the tunnel centerline upstream of the rotor require microphones to be placed sufficiently below the plane of rotation to avoid ingestion of wakes from the microphone support stand. In the plane of rotation measurements are limited to azimuthal angles, ψ, for which the microphone support stand wakes are not convected into the rotor disk. These points are illustrated using a scaled figure based on a study conducted in an existing facility. On the downstream side, microphones must be placed outside the rotor wake to avoid sensing the pressure fluctuations in the downwash flow.

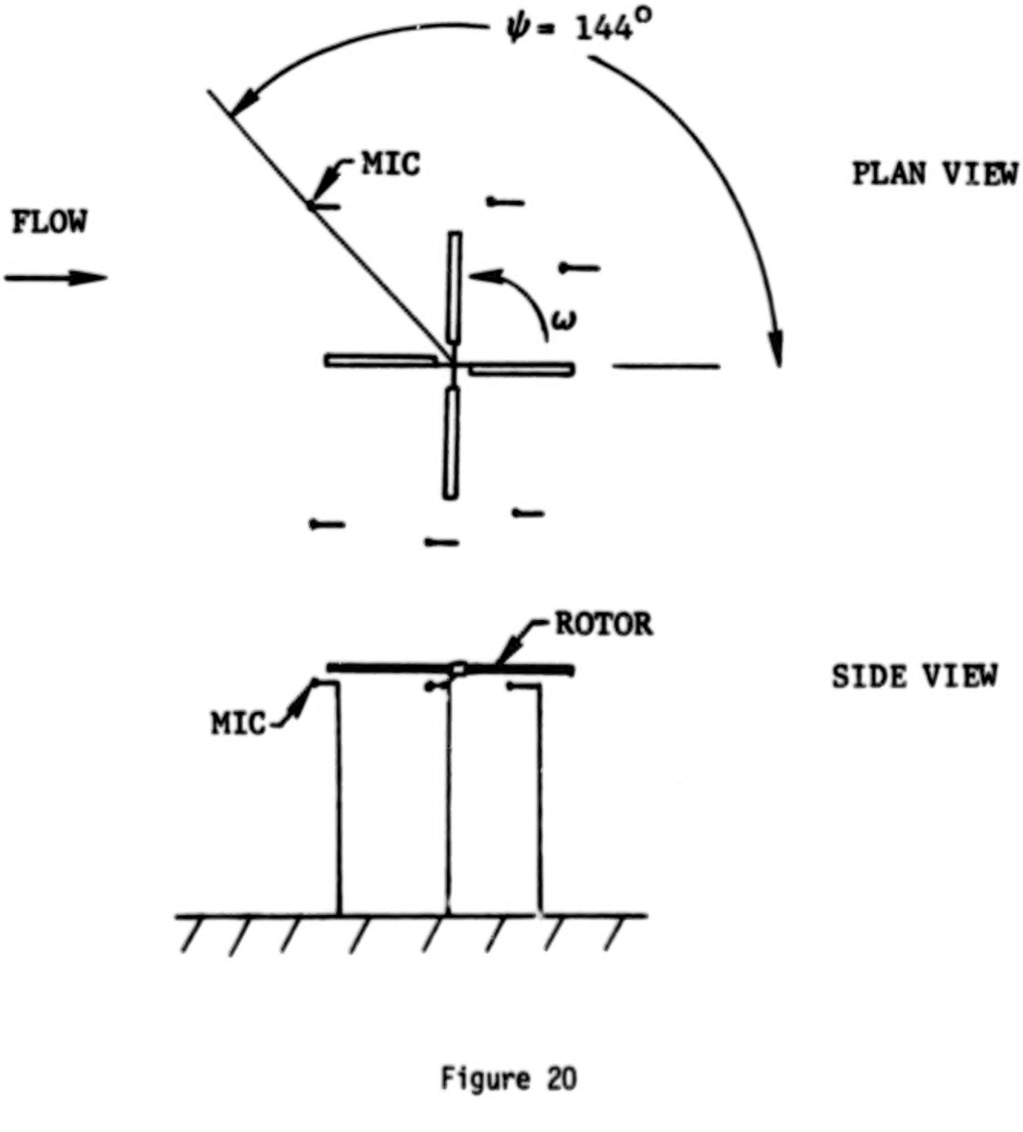


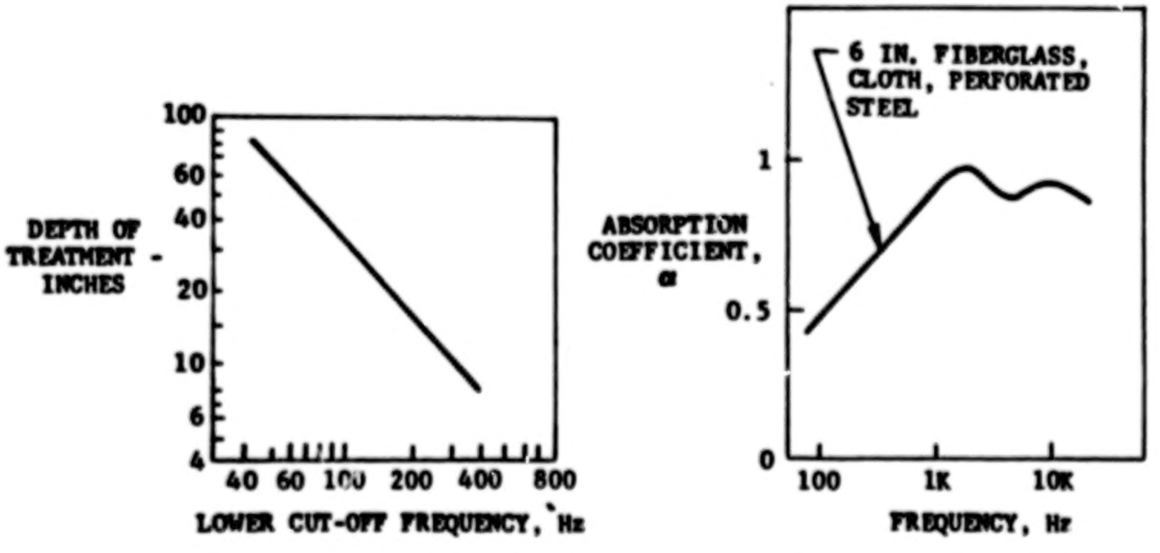
Figure 20

ACOUSTIC LINING ABSORPTION CHARACTERISTICS

Acoustic lining is used in open-jet test facilities and wind tunnels to minimize acoustic reflections. Since the absorption characteristic of the lining varies with acoustic source frequency it is instructive to evaluate the frequency dependence of typical lining materials. In the case of fiberglass wedges, the depth of treatment increases significantly as the cut-off frequency, corresponding to 99% absorption, decreases. The excessively deep wedges needed for frequencies below 150 Hz represent a prohibitive cost and size limitation in many facilities.

If a combination of fiberglass blankets, cloth, and perforated steel is used, which is typical of wind tunnels, then the frequency dependence of the absorption coefficient indicates a 90% absorption at approximately 800 Hz. At 100 Hz the lining has only a 50% absorption. Such a low absorption coefficient represents a limitation when testing full-scale rotors in a wind tunnel. This is because the blade passing fundamental frequency and several high harmonics occur below 100 Hz where the absorption coefficient is low. To ensure accurate free-field data, it is desirable to use smaller rotors although this choice could raise questions about aerodynamic and aeroelastic scaling effects. Also, it is difficult to install surface transducers on smaller rotors due to the small chord and blade thickness.

FUNDAMENTAL FREQUENCY: FULL SCALE 44 FT ROTOR - 5 Hz x N
MODEL SCALE 6 FT ROTOR - 30 Hz x N



FIBERGLASS WEDGE DESIGN DIMENSIONS

ACOUSTIC LINING IN TYPICAL WIND TUNNEL

Figure 21

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UNIFORMITY OF FREE-FIELD DECAY IN MULTIPLE DIRECTIONS

The uniformity of the free-field decay characteristics in an acoustic facility influences the quality of the source directivity data. Ideally if the decay curves in all directions are plotted on the same figure, the scatter should be ±1dB about the inverse square law decay curve. This gives a ±1dB uncertainty in the measured acoustic source directivity pattern. Such an acoustic performance requirement is not easily attained as shown by the large scatter occurring at 250 Hz and 8 KHz in a typical wind tunnel with acoustic lining.

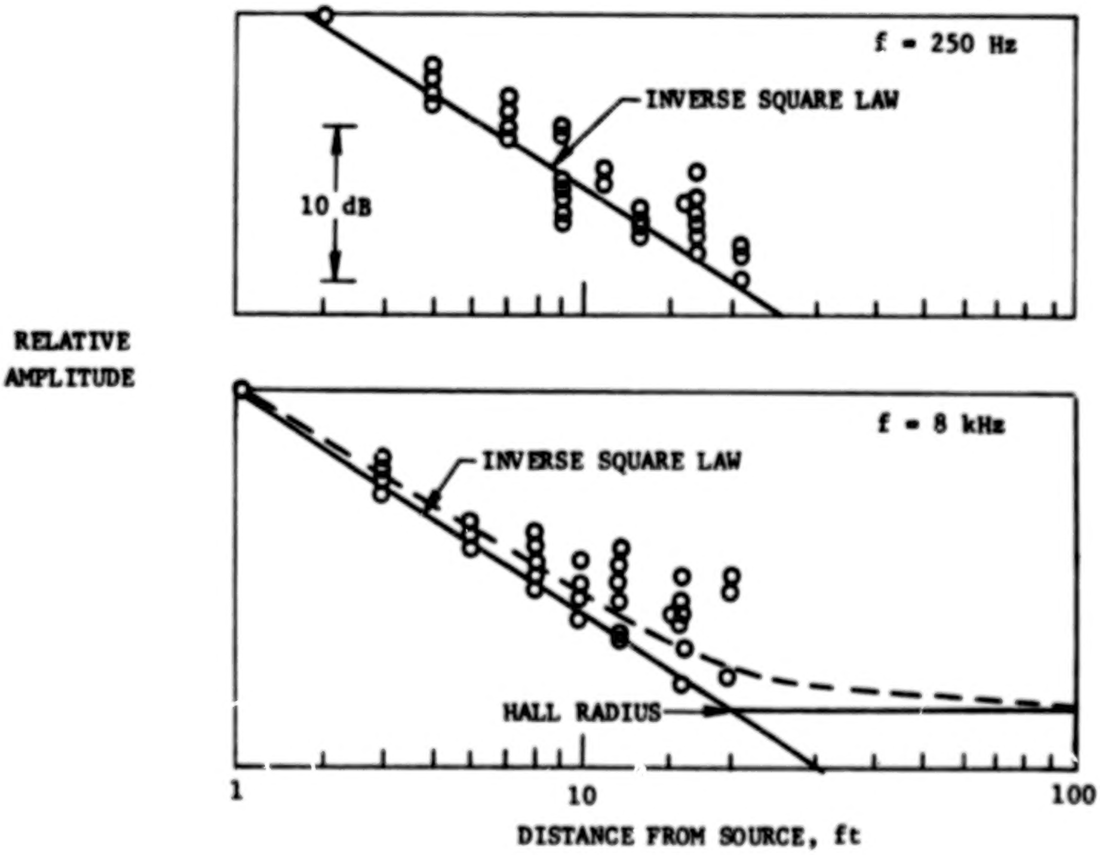

Figure 22

SOURCES OF BACKGROUND NOISE

In addition to microphone self noise, facility background noise sources can limit the signal-to-noise ratio in an experiment. The sources of this extraneous noise vary from the sound generated by the tunnel drive system to noise produced by flow over perforated acoustic liners. Often the background noise is a function of the microphone position. For example, this noise is stronger at downstream stations in an open-jet facility due to impingement of the flow on the collector.

The background noise is also expected to dominate over the high-frequency broadband noise generated by rotors. This postulation is based on the need to use sophisticated source location techniques (directional microphones, cross-correlations) in recent studies of isolated airfoil trailing-edge noise (Ref. 3) in an open jet.

It should be noted that a large signal-to-noise ratio does not ensure a free-field condition for acoustic measurements. This is because a reverberant field could still exist during the individual background noise and rotor noise measurements used to determine the signal-to-noise ratio.

- **TUNNEL DRIVE SYSTEM**
- **STALLED FLOW IN TUNNEL CIRCUIT**
- **OPEN-JET SHEAR LAYER**
- **OPEN-JET IMPINGEMENT ON COLLECTOR**
- **SUPPORT STANDS, STINGS**
- **ROTOR DOWNWASH IMPINGING ON FLOOR**
- **WALL REFLECTIONS IN CLOSED WIND TUNNELS**
- **FLOW OVER PERFORATED LINERS USED AS ACOUSTIC TREATMENT**

Figure 23

INFLUENCE OF SUPPORT STING ON BACKGROUND NOISE

It is important to simulate the complete test system, with the exception of the rotor, when conducting background noise measurements. This is illustrated by two background noise calibrations obtained with and without a sting used to support test vehicles in an open-jet acoustic wind tunnel. Differences between the background noise spectra indicate significant noise can be generated by facility hardware or possibly a fuselage used to simulate a complete helicopter installation.

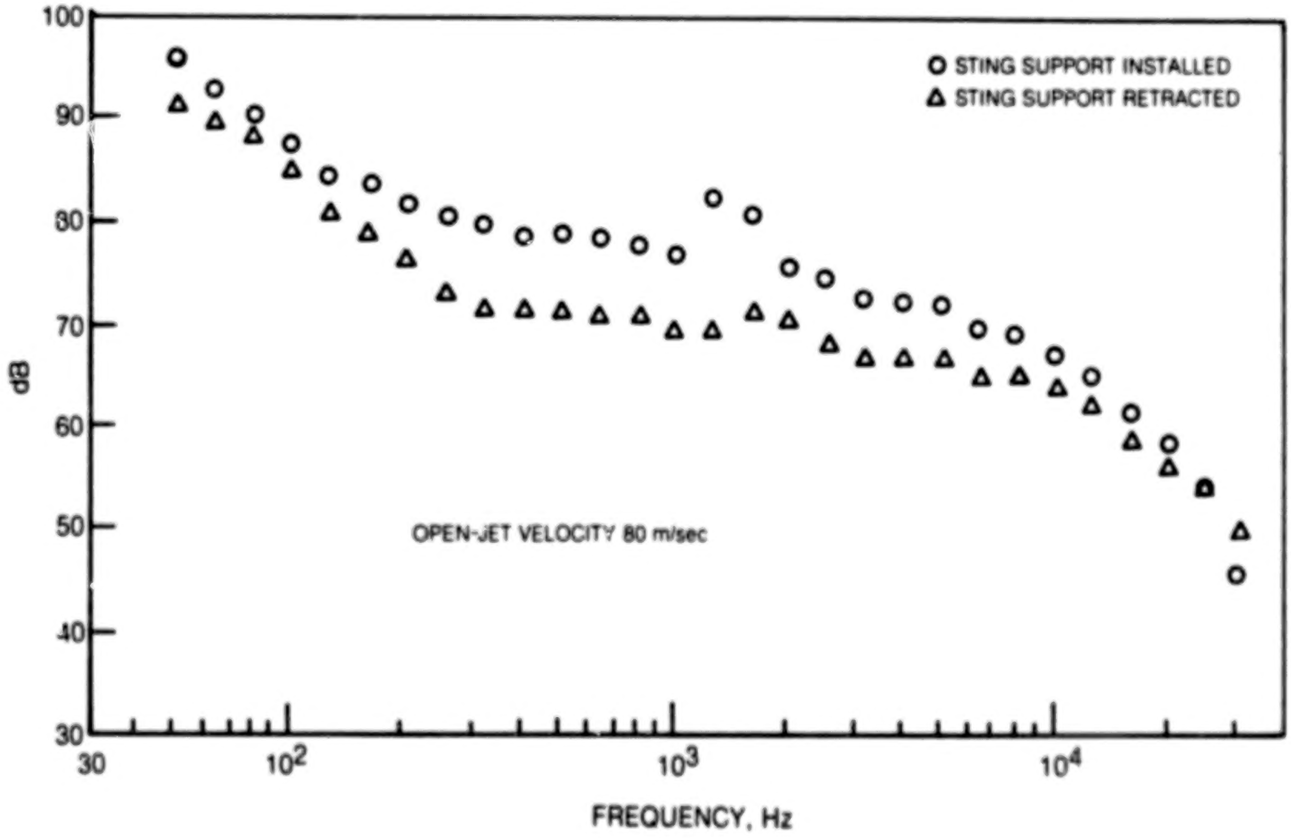

Figure 24

FLYOVER TESTING METHOD

The previous figures concentrated on ground-based test facilities. Full-scale flyover testing can also be used to generate a helicopter noise data base. This approach provides a direct assessment of the helicopter noise versus certification criteria. However, such data is difficult to use in assessment of prediction methods for the different helicopter noise sources since all sources occur simultaneously. In addition, ground reflections and moving source effects complicate interpretation of the data in comparisons with first principle analyses. These observations should not preclude conducting future flyover tests as discussed later in a summary of existing facility capabilities.

Several technical details must be considered when conducting flyover tests. These range from the effect of wind gusts on the noise generation to uncertainties in the rotor operating condition. Conducting accurate acoustic tests requires minimizing these effects.

- **Technical considerations**
    Wind/gust effects on:
        Intermittent sources near M=1 (Ex. BVI)
        Cross flow into rotor
        Vortex path changes in BVI
        Turbulence ingestion noise
        Flight path changes
    Uncertainties in acoustical transmission path
    Attenuation due to humidity (significant)
    Aircraft position uncertainty
    Rotor operating condition uncertainty (Ex. thrust)

Figure 25

IN-FLIGHT TESTING METHOD USING SECOND AIRCRAFT

While flyover tests are limited mainly to product assessment, in-flight testing using a second aircraft can be applied to basic research programs. The advantage of this approach is that it provides nearly free field directivity data. In addition, complex manuevers such as descent (which results in blade-vortex interaction) can be investigated. The success of the technique is based on the occurrence of a single high-intensity acoustic source. In this case, the high-intensity source dominates over the remaining sound sources permitting diagnostic tests to be conducted using acoustic signatures in the time domain. Weaker sources are not easily identified unless they are deterministic and easily distinguished in the time domain. Under these conditions, signal enhancement can be used to extract the weaker sound source. In contrast, rotor broadband noise sources cannot be identified using such techniques. It is presently postulated that these sources would be dominated by the self noise on the in-flight microphone in addition to engine noise and possibly airframe noise.

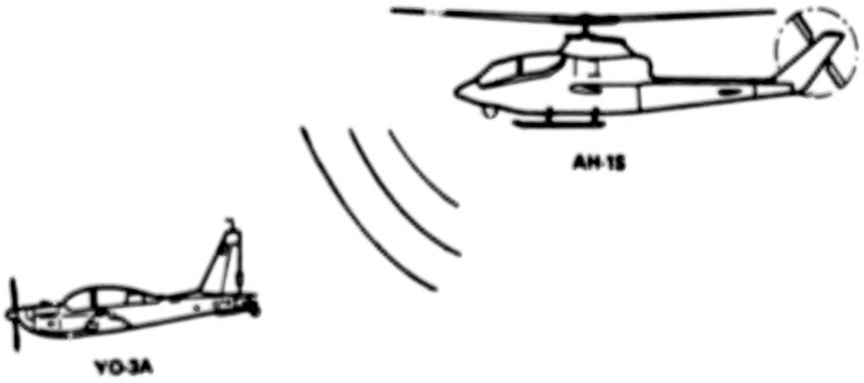

Figure 26

SUMMARY OF FACILITY CAPABILITIES IN THE UNITED STATES

This presentation has emphasized facility criteria for obtaining an accurate helicopter data base. These criteria can now be compared with existing facility capabilities. In the case of open-jet acoustic wind tunnels, many of the aerodynamic and acoustic performance requirements described earlier can be satisfied. Such facilities provide experimental data directly applicable to basic research programs.

The size of the open-jet test section in existing facilities restricts studies to model rotors approximately 1 meter in diameter. The resulting small blade chord limits these studies to acoustic sources which are Reynolds number independent. It still remains to verify that data obtained in such studies can be extended to the full-scale flyover conditions. It should be noted that Reynolds number dependence can be investigated for selected acoustic sources, such as trailing-edge noise, using two-dimensional sections of a full-scale rotor blade (Ref. 3).

The rotor diameter limitation encountered in open jets can be circumvented by testing a larger rotor in wind tunnels. Such test facilities represent a potential basic research capability. However, the aerodynamic and acoustic performance capabilities of existing facilities must be documented to determine if they satisfy the requirements delineated here. In a few cases improvement of the flow and acoustic properties is needed.

A small rotor with approximately a 2 or 3 m diameter may provide the best simulation technique for use in existing wind tunnels. For example, placement of such a rotor in the NASA Ames 12 x 24 m tunnel would ensure that the measurement microphones are in the geometric and acoustic far field of the source. In addition, the frequency range of the resulting spectrum would coincide with the optimum range of the acoustic lining absorption characteristics.

In comparison, the low acoustic source frequencies associated with full-scale rotors (for example 12 m) could limit such studies to relative noise level comparisons for design optimization. It is difficult in this case to conduct diagnostic tests directed towards isolating and understanding the low-frequency contributions to the noise due to the acoustic lining limitations. It is recognized, however, that the idea of testing smaller rotors could be limited by the facility signal-to-noise ratio.

Flyover tests provide direct certification measurements and represent a valuable data base for designing derivative aircraft. Such measurements are presently difficult to apply when assessing the noise generated by individual source mechanisms. This is due to the simultaneous noise generation by all source mechanisms in addition to moving source and ground reflection effects. These restrictions will be removed when sophisticated prediction procedures are available to predict all source contributions to the acoustic spectrum.

Conceptually, contributions from each source would be superimposed to predict the total acoustic spectrum sensed by a ground-based observer. In addition, ground reflections and moving source effects would be included. Once this capability exists, benchmark-quality flyover measurements will become critical to the assessment of the total aircraft prediction method. However, until this situation occurs, assessment of prediction methods for individual noise mechanisms should rely on simulation tests which isolate the desired acoustic source.

In-flight measurements using a second aircraft represent one successful approach to isolating select acoustic sources in the absence of ground reflection and moving source effects. Presently this method can only be applied when a single deterministic noise source clearly dominates over all other noise mechanisms.

CONCLUDING REMARKS

Aerodynamic and acoustic performance criteria have been identified for ground-based facilities needed to generate an accurate helicopter noise data base. Documentation for existing facilities in the United States indicates that open-jet acoustic wind tunnels and acoustically lined hover stands satisfy many of the performance requirements. Rotor models tested in acoustic wind tunnels may, however, be limited in size, thereby restricting their application to Reynolds-number-independent acoustic source mechanisms. Further documentation is needed for wind tunnels before performance capabilities and limitations of this particular testing method can be compared with the criteria described here.

The present criteria can be used to identify future facility improvements needed to achieve state-of-the-art test methods in existing facilities within the United States. A good example of a multipurpose facility which considers these criteria is the DNW open-jet acoustic wind tunnel in The Netherlands.

Flyover testing methods can also provide an experimental rotor noise data base. Measurements can be used to assess certification standards and plan derivative aircraft. In-flight measurements using a second aircraft represent a viable approach to investigating clearly dominant full-scale rotor noise sources.

REFERENCES

1. Paterson, R. W.; and Amiet, R. K.: Noise of a Model Helicopter Rotor Due to Ingestion of Turbulence. NASA CR-3213, 1979.

2. Schlinker, R. H.; and Amiet, R. K.: Refraction and Scattering of Sound by a Shear Layer. NASA CR-3371, 1980. (Paper also presented at AIAA 6th Aero-acoustics Conference, June 4-6, 1980, Hartford, Conn., AIAA Paper No. 80-0973.)

3. Schlinker, R. H.; and Amiet, R. K.: Helicopter Rotor Trailing Edge Noise. NASA CR-3470, 1981. (Paper also presented at AIAA 7th Aeroacoustics Conference, October 5-7, 1981, Palo Alto, Calif., AIAA Paper No. 81-2001.)

N84 29673

High-Speed Noise of Helicopter Rotors

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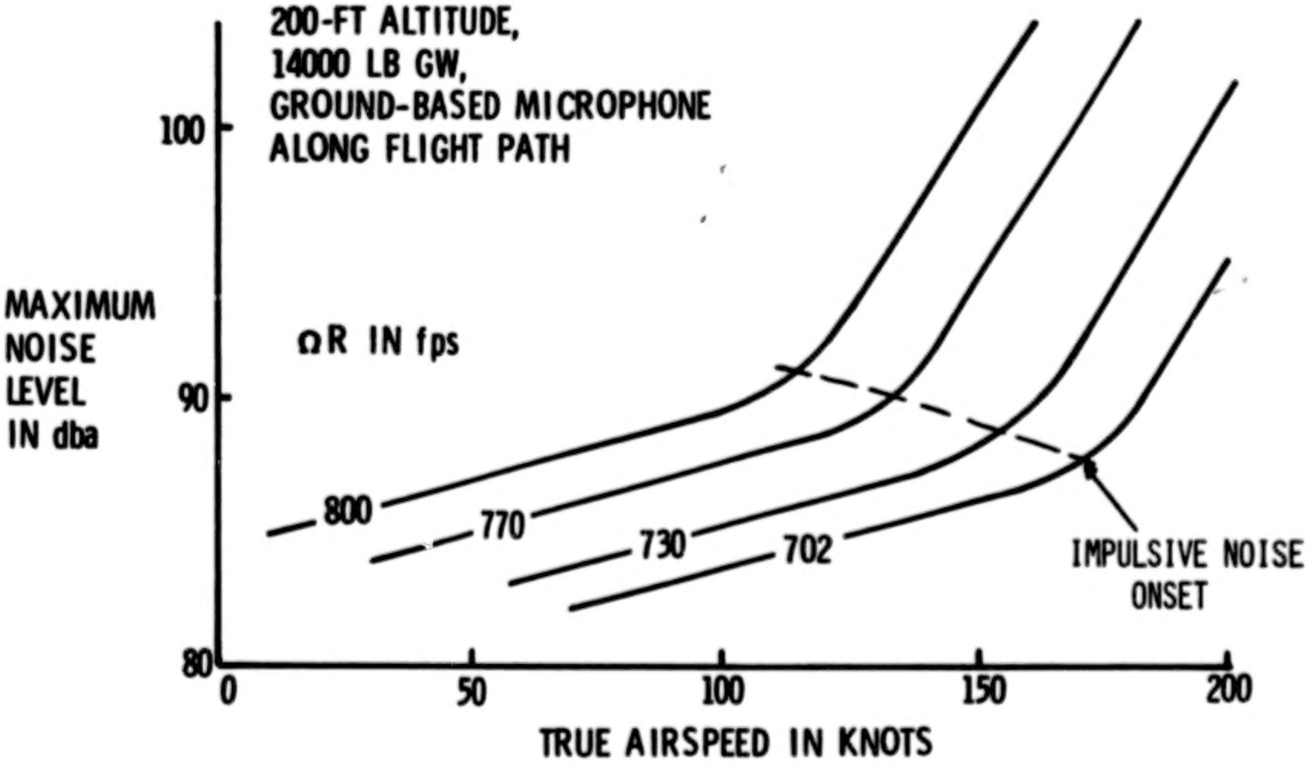
Flyover Noise Trends of Helicopter Rotors.

Flyover noise trends of helicopter rotors at 200-ft altitude are shown below. The abrupt increase in the measured levels indicates the onset of impulsive noise. Similar trends can also be seen from in-plane noise measurements (Ref. 1 and 2).

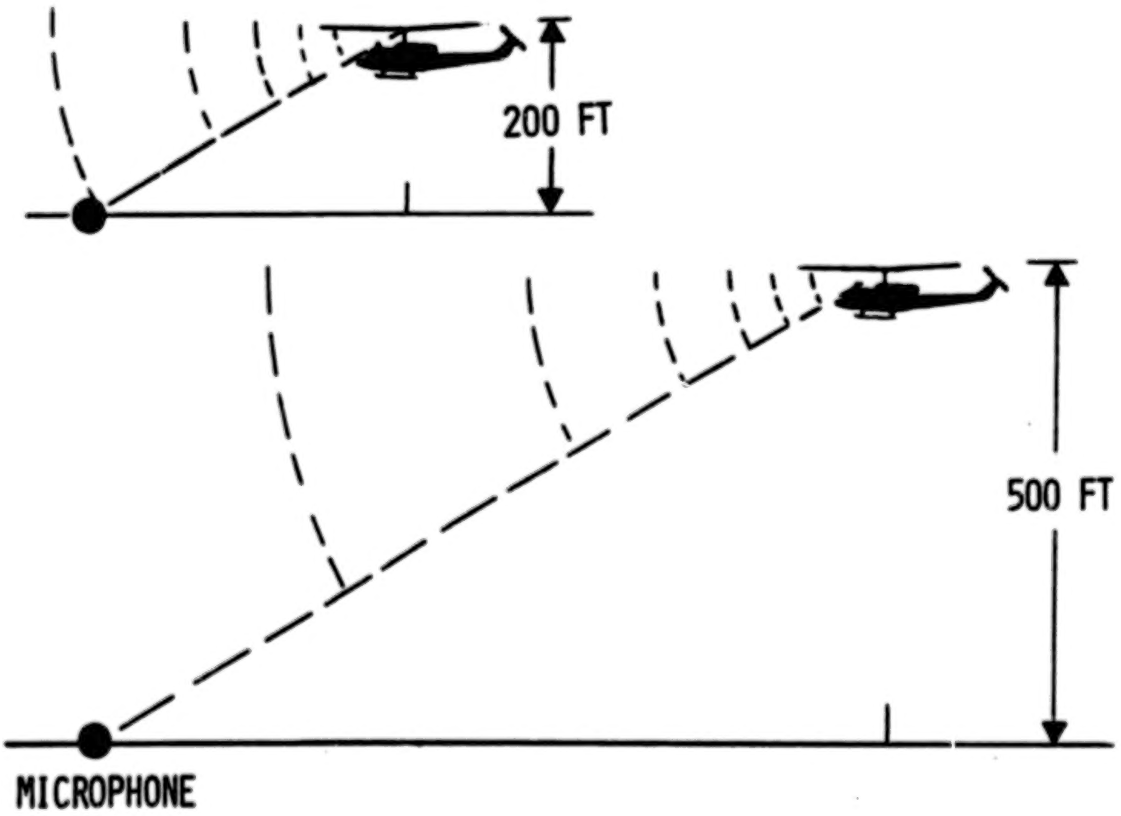
NOISE LEVELS INCREASE ABRUPTLY AT IMPULSIVE NOISE ONSET



Effect of Directivity on Measured Noise.

High-speed impulsive noise is highly directional in the plane of rotation (Ref. 1). As shown in the figure below, the impulsive noise at higher altitudes is recorded when the helicopter is farther from the microphone. This results in a reduction in the measured noise. Hence, an abrupt increase in noise levels may not occur for high-altitude flyovers.

ABRUPT INCREASE IN NOISE IS NOT PROMINENT FOR 500-FT FLYOVER

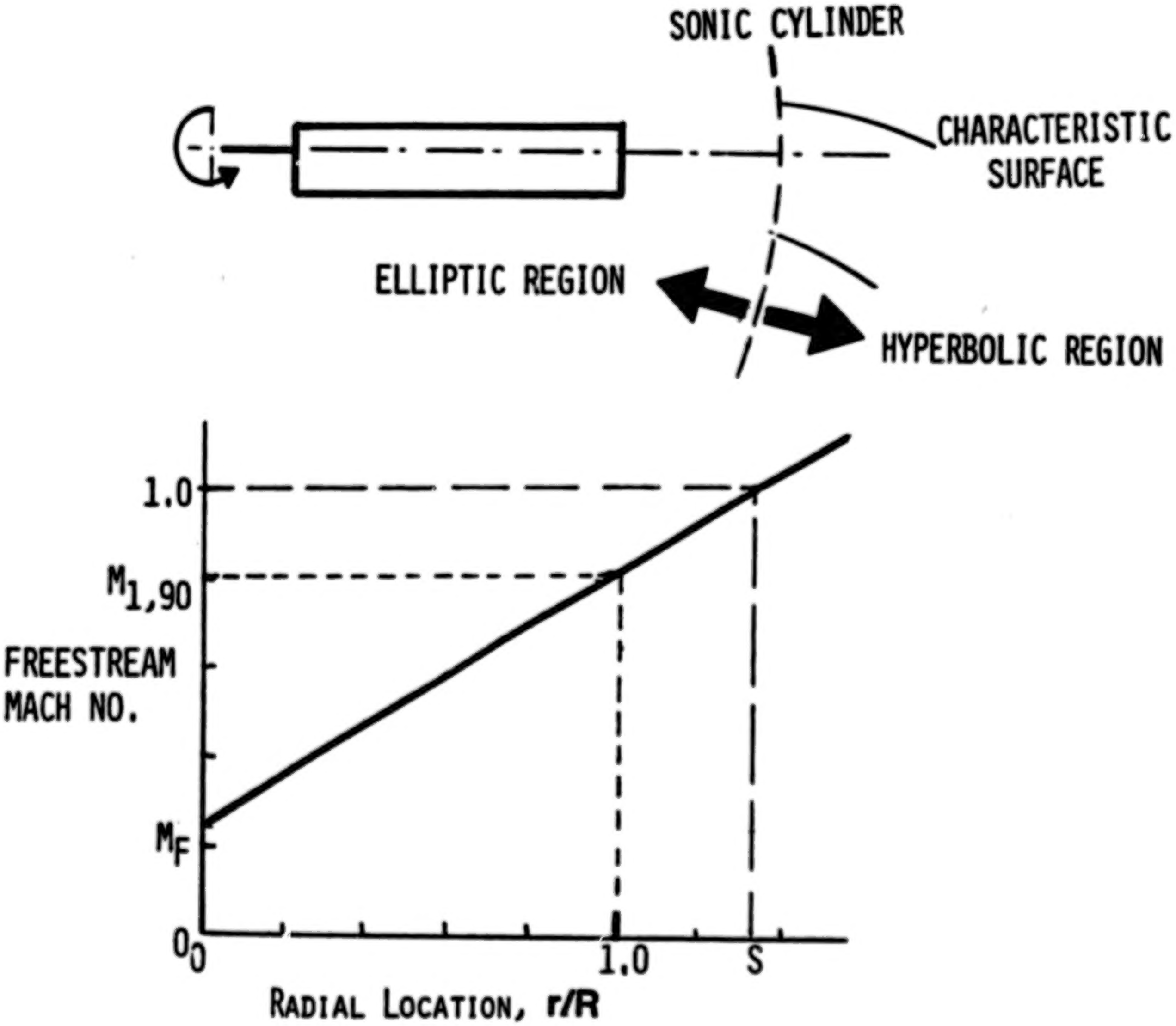


Effect of the Flow Regions on Noise.

Understanding the properties of the flow regions of a helicopter rotor sheds light on the phenomenon of impulsive noise (Refs. 3 and 4). In brief, disturbances within an elliptic region are felt in the entire region. In contrast, they are propagated only along surfaces of characteristics in a hyperbolic region. This makes the hyperbolic region a better medium for intensely directed disturbance propagation.

Both regions exist in the flow field of a rotor. A cylindrical surface called the sonic cylinder (Ref. 4) separates these two regions. The Mach number is unity at the sonic cylinder.

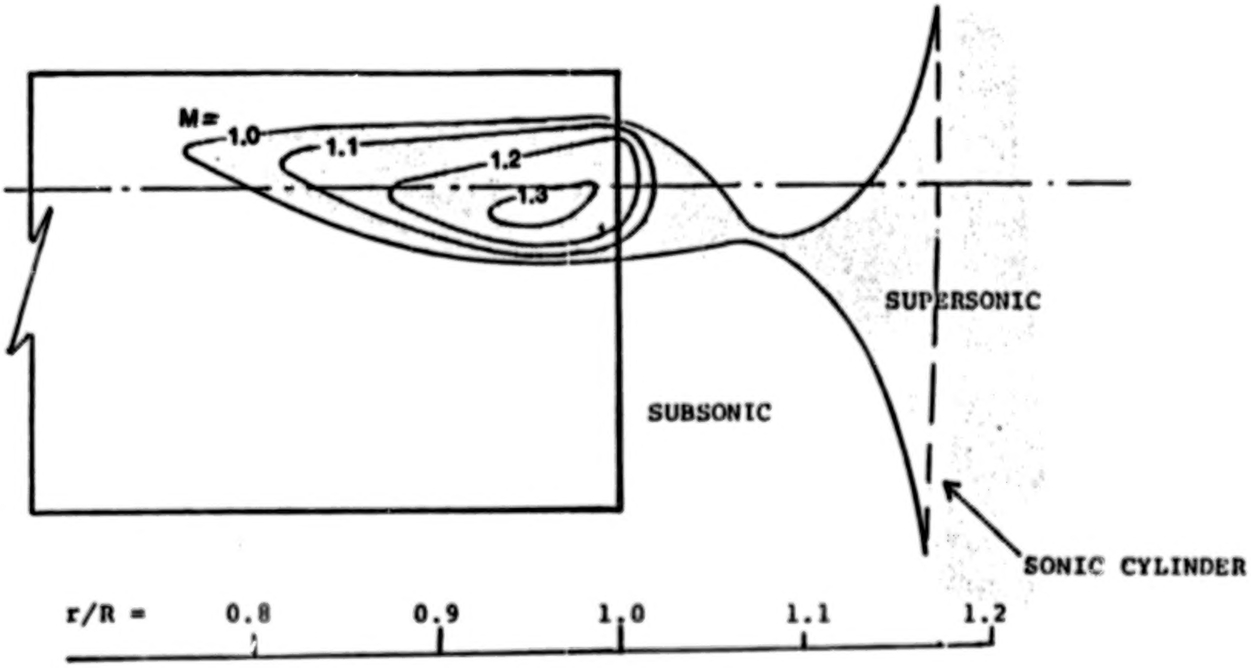
SONIC CYLINDER SEPARATES HYPERBOLIC AND ELLIPTIC REGIONS


SONIC CYLINDER
CHARACTERISTIC SURFACE
ELLIPTIC REGION
HYPERBOLIC REGION
1.0
$M_{1,90}$
FREESTREAM MACH NO.
M_F
0
0
1.0
S
RADIAL LOCATION, r/R

Delocalization and Impulsive Noise.

At transonic speeds, a supersonic region develops near the advancing blade tip and we will refer to it as the inner hyperbolic region. At large enough speeds, this region merges with the hyperbolic region outside the sonic cylinder forming an uninterrupted hyperbolic region from blade tip to the far field. This phenomenon is called delocalization (Ref. 4) and it represents the onset of impulsive noise. Therefore, the sonic cylinder radius (S) and the inner hyperbolic region are the key factors which determine the propagation medium.

DELOCALIZATION OCCURS AT LARGE ENOUGH TIP SPEED



Universal Noise Curves.

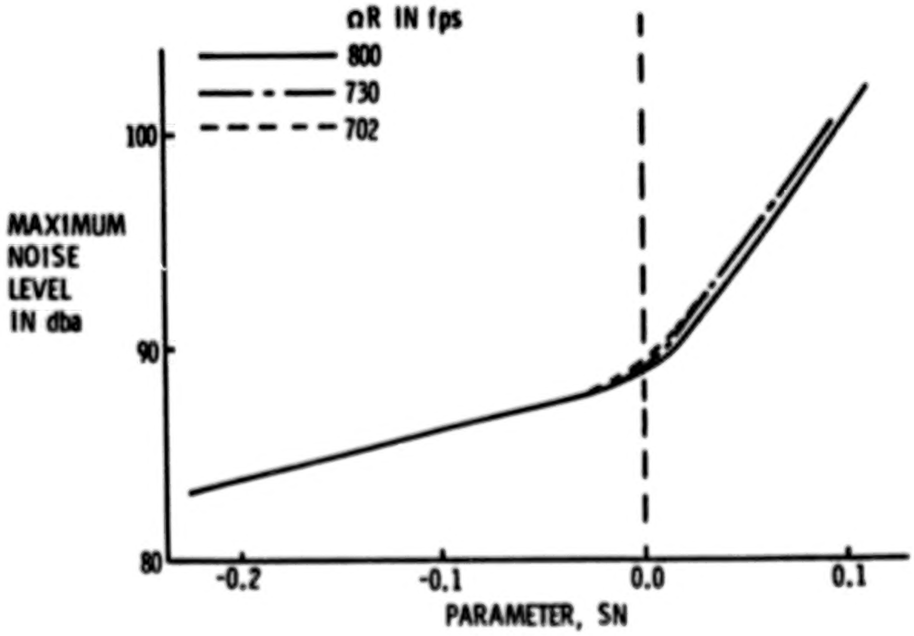
At the onset of inpulsive noise, the inner hyperbolic region just merges with the external hyperbolic region. Therefore, the sonic cylinder radius at this onset (S_s) provides an approximation for the extent of the inner hyperbolic region. Then one can define a new parameter as

$$SN = S_s - S$$

Use of parameter SN with empirical relationships to account for changes in gross weight, blade chord, and rotational tip speed collapses a class of noise data onto a universal curve. Such a noise curve is derived from the 200-ft flyover data shown earlier and presented below.

Note: Due to the complexities of the operating environment, tail rotors do not behave consistently. In addition, available data is very limited for the tail rotors.

PARAMETER SN WITH EMPIRICAL RELATIONS
PROVIDES UNIVERSAL NOISE CURVE



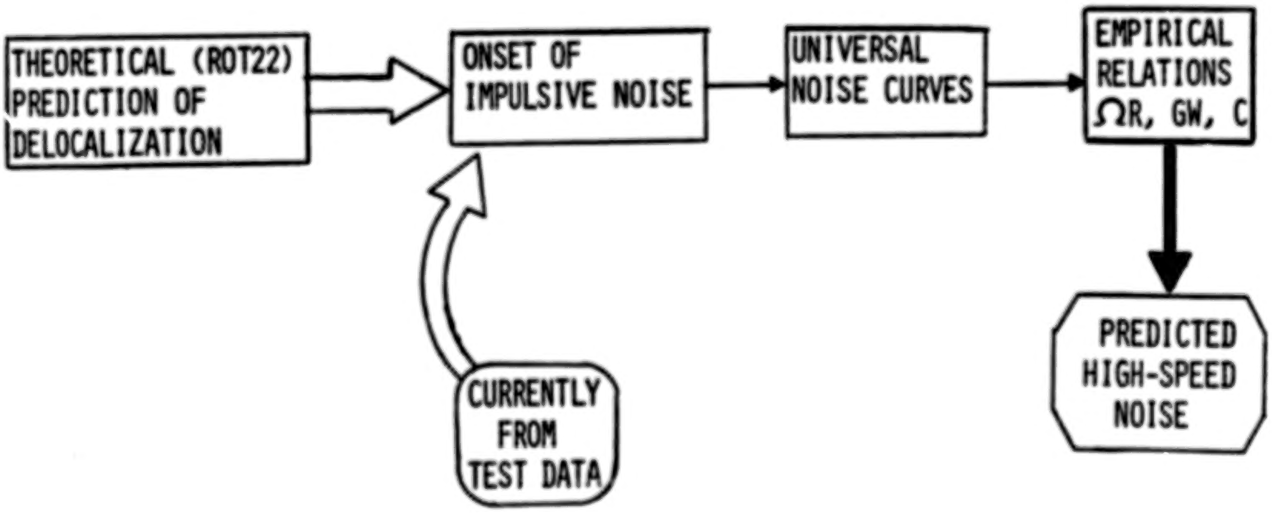
Application of Universal Curve for Noise Prediction.

Universal noise curves can be used for noise prediction if the impulsive noise onset condition is known. Currently it is estimated from test data. However, it is desirable to be able to theoretically predict the onset of impulsive noise. The diagram, shown below, indicates how this task can be accomplished.

ROT22 is a 3-D, quasi-steady, full-potential, transonic-rotor-blade analysis program developed by NASA (Ref. 5). It is hoped that this program will provide an insight into the effects of various design parameters on high-speed noise. Preliminary results of such investigations are discussed next.

UNIVERSAL NOISE CURVE CAN BE USED FOR NOISE PREDICTION IF ...

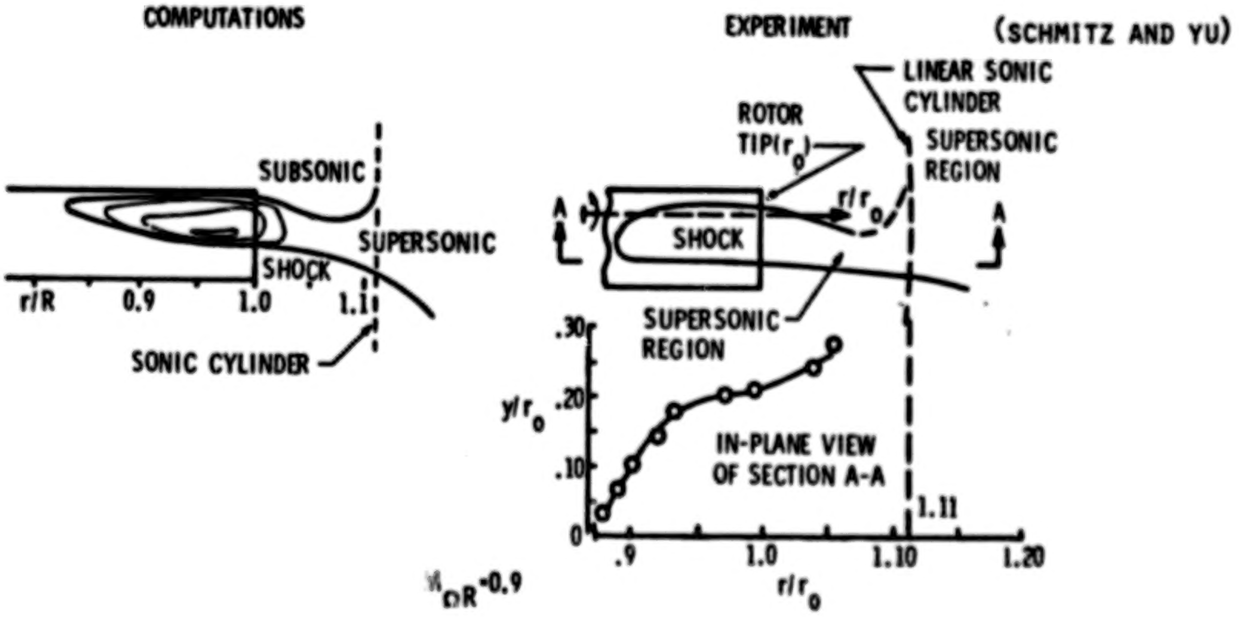
IMPULSIVE NOISE ONSET CONDITIONS ARE KNOWN.


THEORETICAL (ROT22) PREDICTION OF DELOCALIZATION
ONSET OF IMPULSIVE NOISE
UNIVERSAL NOISE CURVES
EMPIRICAL RELATIONS ΩR, GW, C
PREDICTED HIGH-SPEED NOISE
CURRENTLY FROM TEST DATA

Correlation of ROT22 results with Experiments.

The flow field predicted by ROT22 for a 1/7th scale model UH-1H agrees well with the experiments of Ref. 6. Though this agreement is encouraging, experiments should be conducted on different rotor designs and correlated with theoretical predictions.

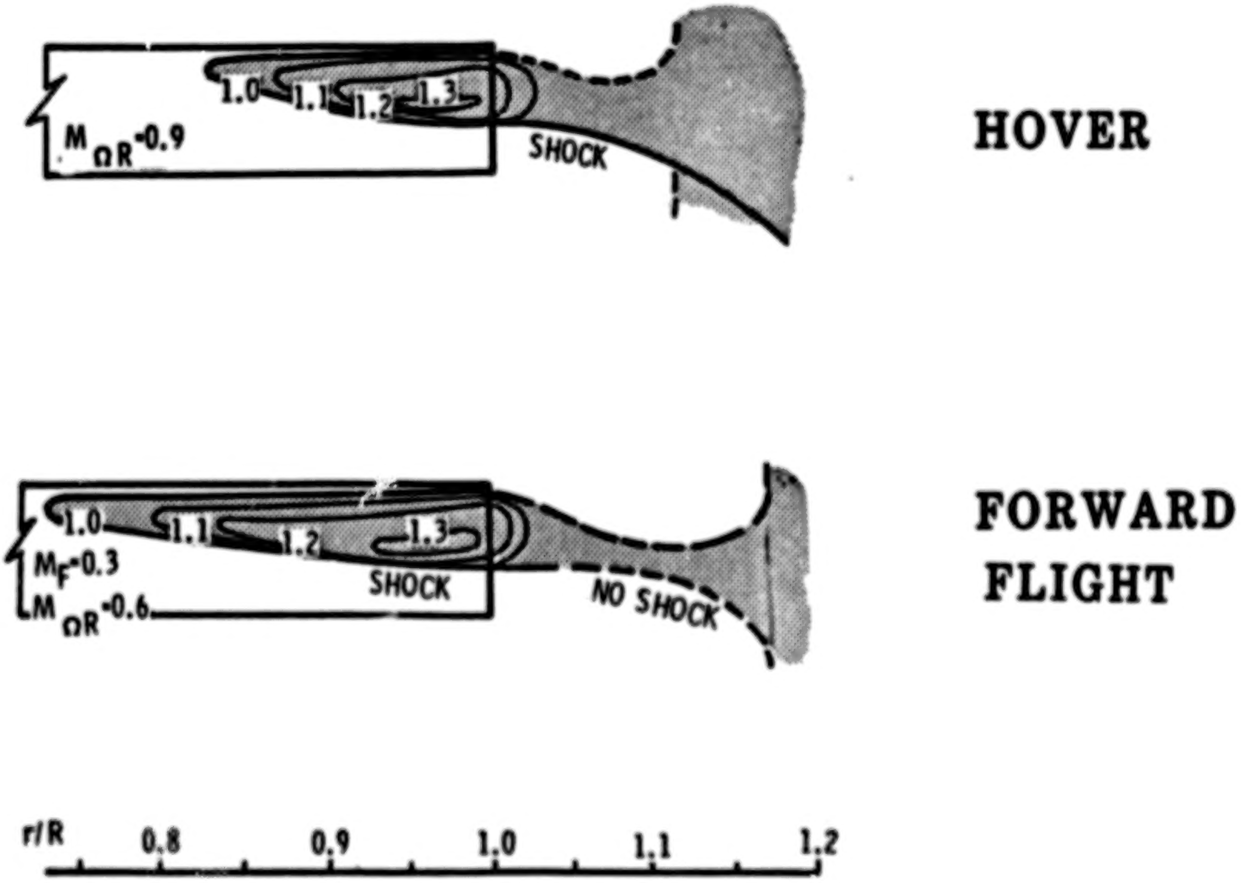
FLOW FIELD PREDICTED BY ROT22 AGREES WITH EXPERIMENTS



Effect of Rotational Tip Speed.

Flow field studies show that the increase in rotational tip speed results in an early onset of impulsive noise. This is because the sonic cylinder is closer to the tip of the blade with higher rotational tip speed.

LARGER ΩR MEANS EARLY IMPULSIVE NOISE ONSET


1.0
1.1
1.2
1.3
$M_{\Omega R}$=0.9
SHOCK
HOVER
1.0
1.1
1.2
1.3
M_F=0.3
$M_{\Omega R}$=0.6
SHOCK
NO SHOCK
FORWARD FLIGHT
r/R
0.8
0.9
1.0
1.1
1.2

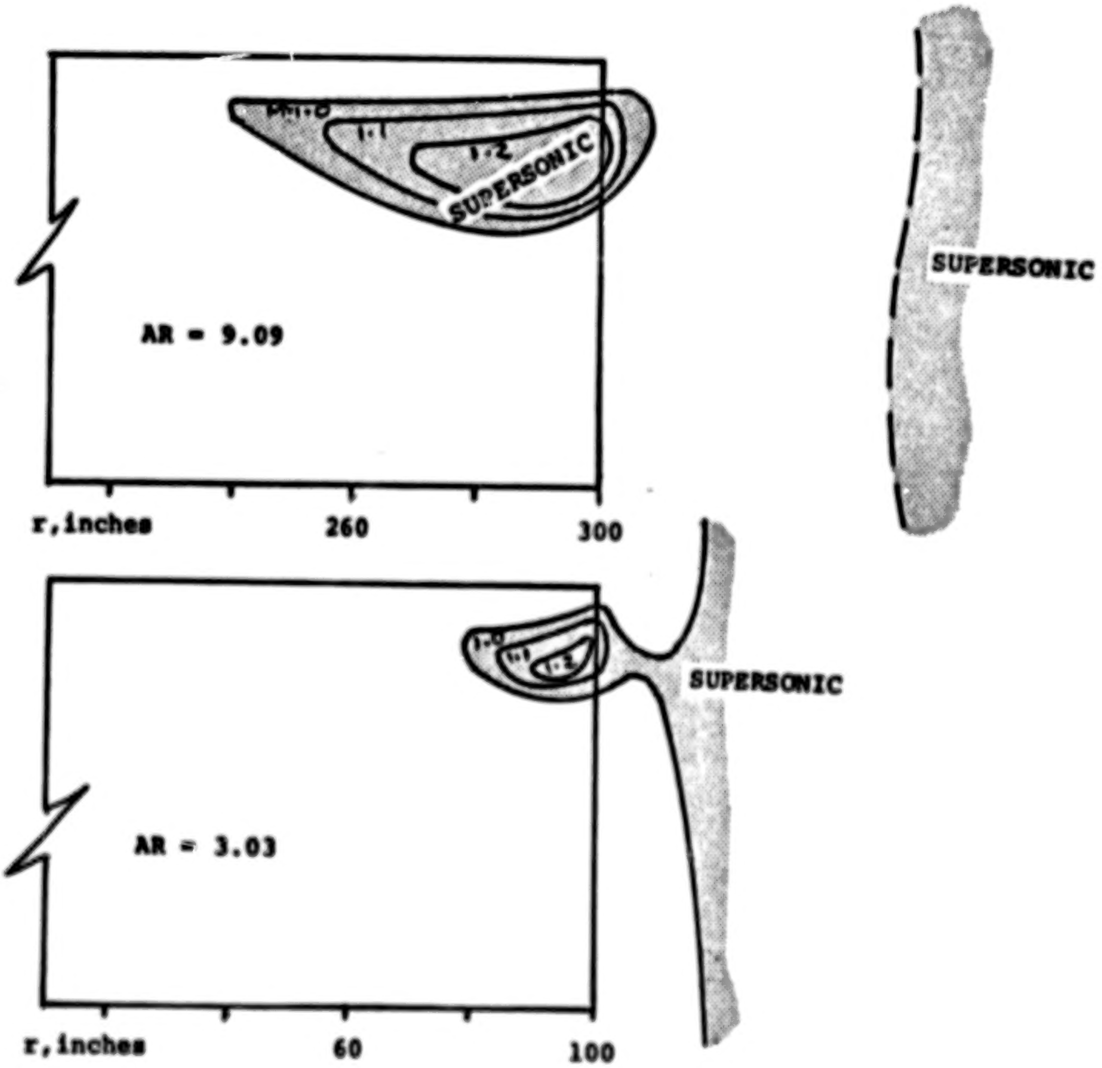
Effect of Aspect Ratio.

Higher aspect ratio (AR) delays the onset of impulsive noise.

A. Radius (r) is varied: The sonic cylinder is physically farther from the blade with larger radius (see figure below). Therefore, the impulsive noise onset is delayed.

B. Chord is varied: The inner hyperbolic region is smaller on the blade with smaller chord (larger aspect ratio). Therefore the onset of impulsive noise is delayed.

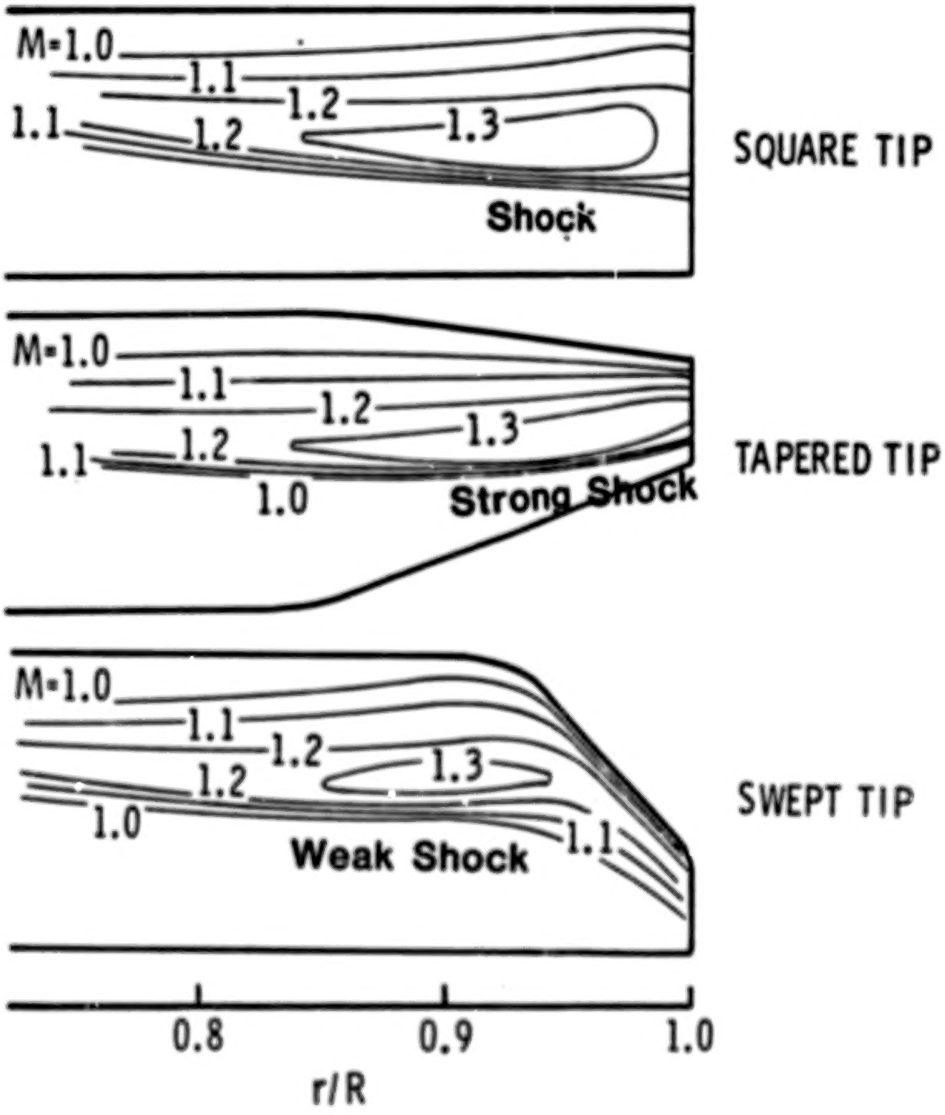
INCREASE IN ASPECT RATIO DELAYS DELOCALIZATION



Effect of Tip Shapes.

Proper tailoring of the tip shape reduces high-speed noise. There are two reasons for this effect. One of them is the reduction in the inner hyperbolic region due to reduced tip chord. Both tapered and swept tips provide this benefit. The other reason is the reduction in the compressibility effects. While swept tip provides compressibility relief, a tapered tip increases compressibility as shown below. As a result, the slope of the noise curve for the tapered-tip blade is higher than the same for the square-tip blade. Based on these findings it is concluded that the swept tips are more desirable for high-speed-noise reduction.

PROPER TAILORING OF TIP SHAPES MINIMIZE HIGH-SPEED NOISE



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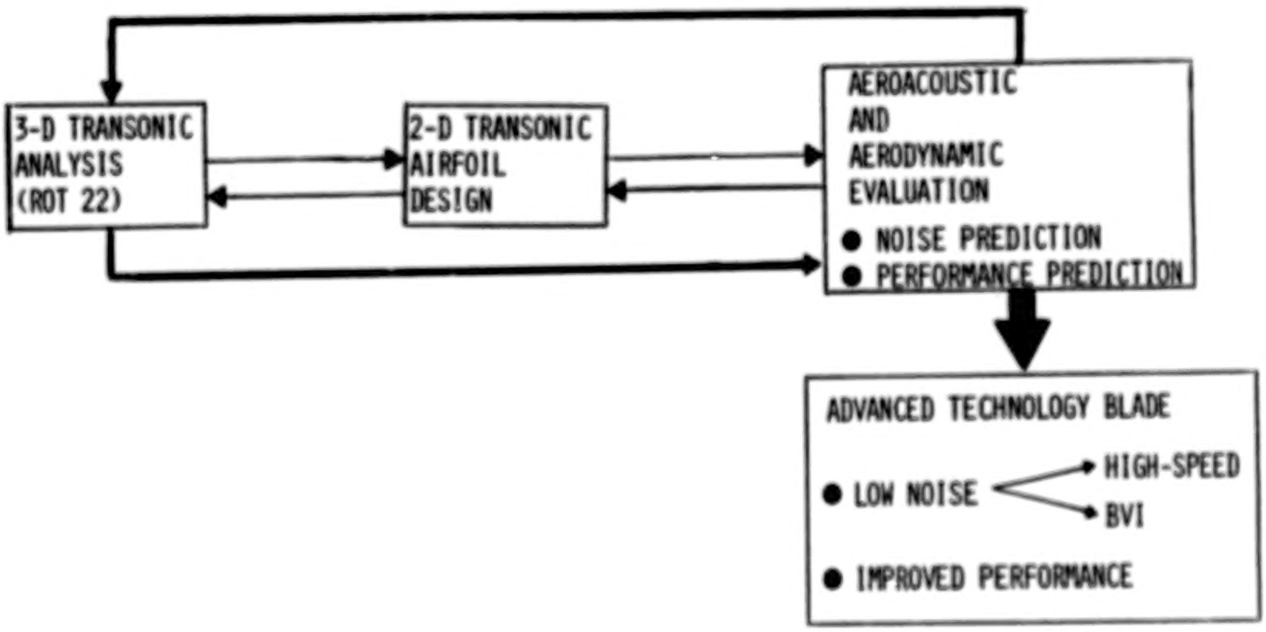
Design of "Low-Noise" Blades.

In addition to noise prediction programs, a feasible design approach must include a variety of aerodynamic programs. Such an approach is shown in the diagram below. The design process can be initiated with a blade designed to meet performance objectives. Then the task will be to reduce high-speed and blade-vortex-interaction (BVI) noise without adversely affecting performance capabilities. As shown below, the 3-D transonic program can be used to identify problem areas. Compressibility effects can be minimized and noise can be reduced by designing new airfoils which meet performance objectives (inner loop). Planform modifications can be made to reduce noise while retaining performance capabilities (outer loop, shown in bold lines). The design process can be terminated when a satisfactory blade is designed.

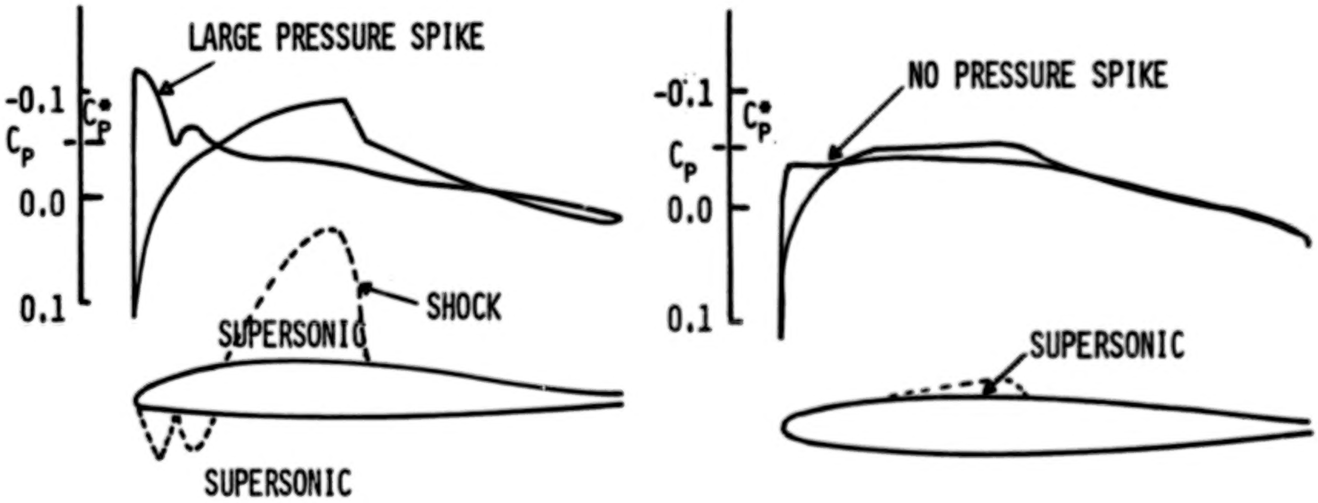
AERODYNAMIC PROGRAMS ARE ESSENTIAL FOR LOW-NOISE BLADE DESIGN


3-D TRANSONIC
ANALYSIS
(ROT 22)
2-D TRANSONIC
AIRFOIL
DESIGN
AEROACOUSTIC
AND
AERODYNAMIC
EVALUATION
● NOISE PREDICTION
● PERFORMANCE PREDICTION
ADVANCED TECHNOLOGY BLADE
● LOW NOISE
HIGH-SPEED
BVI
● IMPROVED PERFORMANCE

Airfoils Designed for Reduced BVI Noise.

It is observed in Reference 7 that most of the fluctuations in the surface pressure due to BVI occurs near the leading edge of the blade. Therefore, airfoils designed to have a gradual increase in the pressure in this area may reduce BVI noise. Experimental verification will be necessary before any final conclusion can be drawn.

PROPER TAILORING OF PRESSURE DISTRIBUTION MAY REDUCE BVI NOISE



$M_\infty = 0.775$

RECOMMENDATIONS

- BROADEN ACOUSTIC DATA BASE
- DEVELOP AND MAINTAIN INTERIM NOISE PREDICTION CAPABILITY
- IDENTIFY FUNDAMENTAL MECHANISMS
- EXTEND AERODYNAMIC CAPABILITIES TO AEROACOUSTIC APPLICATIONS

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N84 29674

ACOUSTIC METHODOLOGY REVIEW

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OBJECTIVES OF THIS PRESENTATION

The objectives of the presentation are shown on the figure below. It is important for industry and NASA to assess the status of acoustic design technology for predicting and controlling helicopter external noise in order for a meaningful research program to be formulated which will address this problem. This paper will address available technology, the impact of this technology to the design process, and those elements of the problem which need to be addressed.

OBJECTIVES

- REVIEW ACOUSTIC DESIGN OPTIONS
- DISCUSS STRENGTHS & WEAKNESSES
- DISCUSS METHODOLOGY REQUIREMENTS

Figure 1

ACOUSTIC PREDICTION OPTIONS

The prediction methodologies available to the designer and the acoustic engineer are three-fold. First is what has been described as a first principle analysis. This analysis approach attempts to remove any empiricism from the analysis process and deals with a theoretical mechanisms approach to predicting the noise. The second approach attempts to combine first principle methodology (when available) with empirical data to formulate source predictors which can be combined to predict vehicle levels. The third is an empirical analysis, which attempts to generalize measured trends into a vehicle noise prediction method. This paper will briefly address each.

ACOUSTIC PREDICTION DESIGN ANALYSIS OPTIONS AVAILABLE

- **1st PRINCIPLE**
- **SEMI-EMPIRICAL**
- **EMPIRICAL**

Figure 2

1ST PRINCIPLE ANALYSIS

Unfortunately, a first principle analysis for helicopter noise prediction which addresses the civil noise problem (mid and high frequency helicopter noise) is not yet available. The requirements for this type of analysis, in order to be useful, are delineated here. Peter Arcidiacono, in his paper, addressed some of the detailed elements which are necessary to formulate such an analysis. The detailed understanding and definition of all of these elements will take years to complete. What is needed first is a better definition of the problem. This involves an in-depth measurement of a typical sampling of current helicopters, which not only separates the source contributions, but also attempts to identify the elements (acoustic content and mechanisms) associated with each source. In this way we can meaningfully prioritize the elements of the program which are in need of the most immediate attention.

As has been recognized, this is not only a problem for the acoustician to solve, but a formidable and challenging one for the aerodynamicist as well. The detail of aerodynamic understanding necessary appears to be an order of magnitude more complex and inclusive than that which exists today. The acoustician must then translate this input into a methematical formulation which transforms that input into an acoustic field. A good start to this mathematical transformation has been made using analysis techniques such as the Nystrom/Farassat analysis.

Once this task is completed, then one must have good models for atmospheric attentuation and ground reflection. This latter element is still being worked. Then, because of the complexity of this approach, one must consider features which will make it attractive and practical for industry to use it, such as correlation and computational efficiency.

The following table lists some of the mechanisms and features which need to be addressed in a first principle analysis. An attempt has been made to prioritize these elements, identify the status of the technology availability, validation and incorporation into the current analysis (Farassat), and address the principal concern(s) with each element.

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1ST PRINCIPLE ANALYSIS

SOURCE	IMPORTANCE	1st LEVEL TECHNOLOGY AVAILABLE	1st LEVEL VALIDATION AVAILABLE	INTEGRATED INTO CIVIL PROGRAM	CONCERNS
1. ROTOR NOISE					
1.1 THICKNESS	1st ORDER	YES	YES	PARTIAL	
1.2 LEADING EDGE					
● TURBULENCE	2nd	PARTIAL	PARTIAL	NO	TRUBULENCE INPUT
● FUSELAGE	1st	PARTIAL	NO	NO	TURBULENCE & LOCATION
● SELF INDUCTION	1st	PARTIAL	PARTIAL	PARTIAL	WAKE GEOMETRY
● TR ON MR	2nd	PARTIAL	NO	NO	
● MR ON TR	1st	PARTIAL	NO	NO	WAKE GEOMETRY
● BLADE RESPONSE	1st	PARTIAL	PARTIAL	PARTIAL	
● SURFACE PRESSURE	1st	YES	YES	PARTIAL	UNSTEADY 3–D EFFECTS
1.3 TRAILING EDGE	1st	YES	PARTIAL	NO	BOUNDRY LAYER MODEL
1.4 QUADRUPOLE	1st	YES	PARTIAL	NO	OFF BODY FLOW FIELD
1.5 TIP VORTEX FORM	2nd	YES	NO	NO	UNSTEADY 3–D EFFECT
1.6 WAKE NOISE	2nd	YES	NO	NO	
2. FUSELAGE	2nd	YES	YES	NO	
3. ENGINE	1st	YES ?	NO	NO	SEMI–EMPIRICAL
4. SPECIAL CONFIG. FEATURES					
4.1 ARBITRARY,MULTIPLE ROTORS	1st	NA	NA	NO	
5. PROPAGATION					
5.1 ATMOSPHERIC	1st	YES	YES	NO	ATMOSPHERIC CHARACTERISTICS
5.2 NONLINEARITIES	1st	YES	NO	NO	VALIDITY OF APPROACH
6. GROUND REFLECTION	1st	YES	NO	NO	GND. IMPEDANCE
7. USER ORIENTED FEATURES					
7.1 PROGRAM AVAILIBILITY	1st	PARTIAL	NO	NA	
7.2 RUNNING TIME	1st	YES	NO	NA	
7.3 COUPLING OF AERO. & ACOST.	1st	PARTIAL	YES	PARTIAL	
PROGRAM	1st				
7.4 ARBITRARY FLIGHT PATH	1st	YES	NO	YES	
7.5 ARBITRARY OBSERVER POS.	1st	YES	NO	YES	
7.6 PERTINENT OUTPUT	1st	YES	NO	NO	
7.7 USER'S MANUAL		YES	–	NO	
7.8 DOCUMENTATION		YES	–	NO	
7.9 PERTINENT CHECK CASE		YES	–	NO	
7.10 PERTINENT VALIDATION CASE		YES	–	NO	

Figure 3

CORRELATION OF FIRST PRINCIPLE ANALYSIS

The chart below shows an example of the current Nystrom/Farassat analysis correlation with measured CH-53A rotor noise. As can be seen, the analysis is very sensitive to the aerodynamic input model used. Also evident is the need for correlation at the higher frequencies of importance to the civil problem.

WAKE GEOMETRY ASSUMPTIONS STRONGLY INFLUENCE RESULTS

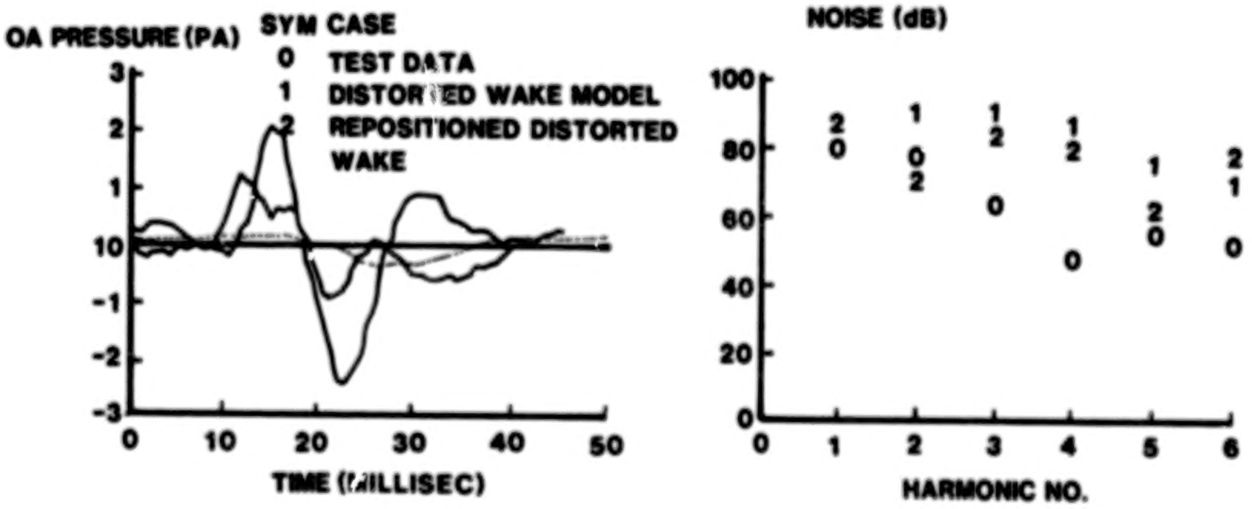


Figure 4

APPLICATION EXAMPLE OF CURRENT 1ST PRINCIPLE ANALYSIS CAPABILITY

Even though the NASA 1st principle analysis of Farassat and Nystrom has some substantial limitations, it does allow a qualitative evaluation of the low frequency harmonic noise for detailed blade design changes. This analysis was used during Sikorsky's recent NASA contract for the design analysis of an Advanced Flight Research Rotor (AFRR). The results of this application are shown on the chart below.

OVERALL SOUND PRESSURE LEVELS FOR FOUR TIPS

	OVERALL	THICKNESS	LOADING
STRAIGHT TIP	87.2	87.1	71.1
BLACK HAWK TIP	86.9	86.8	72.0
BLACK HAWK TIP WITH ADVANCED AIRFOILS	85.1	84.9	71.0
LARGE SWEPT TAPERED TIPS WITH ADVANCED AIRFOILS	82.0	81.7	70.5

Figure 5

1ST PRINCIPLE ANALYSIS RECOMMENDATIONS

If one were to study in depth and develop a detailed analysis for every mechanism of potential importance to the helicopter noise problem, we would probably still be studying the problem when most of us retire. Now that may be an excellent strategy for job security, but I don't believe it is the strategy for industry to rapidly move forward on the problem. What is needed first, as is illustrated in the following figure, is to reach a consensus on the analysis structuring and elements which are required to address the civil (as well as the military) problem. Once these tasks are completed, we must establish goals and dates for these goals. It is important in establishing these goals to define the current accuracy of the best technology available today. This can only be done if we pull together our best analytical elements (whether pure first principle or partly semi-empirical) into a single analysis and correlate this analysis against a carefully measured data base. Only when we know where we are can we then define where we should be going and assign more accurate target dates to our future tasks.

- **REACH CONSENSUS ON STRUCTURING OF THE COMPLETE CIVIL PROBLEM**
- **REACH CONSENSUS ON IMPORTANT ELEMENTS REQ'D**
- **SELECT THOSE FOR INCORPORATION IN A "MARK I" PROGRAM**
- **DEVELOP & RELEASE "MARK I" PROGRAM BY AN AGREED UPON DATE**
- **EVALUATE ACCURACY & CONTINUE TO UPDATE AGAINST ESTABLISHED GOALS**

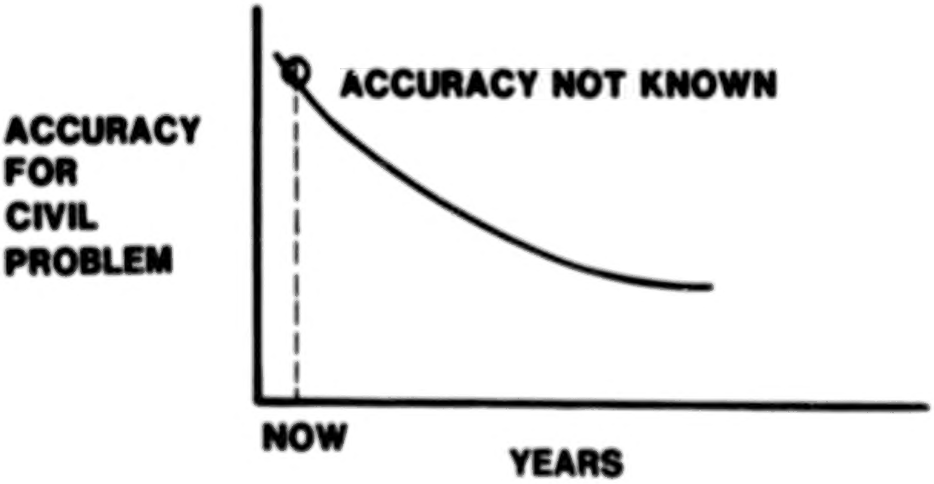


Figure 6

DATA ACQUISITION AND ANALYSIS FOR PROBLEM DEFINITION

There are several facilities and programs which offer the capability to conduct the type of controlled experiments needed to better define the problem, as well as to investigate the mechanisms once the problem definition and prioritizing are accomplished. In order to clearly separate the sources and mechanisms, several candidate representative helicopters will require some detailed acoustic study. Because of the problems associated with translating moving source/ground based data into stationary free field data, aircraft like NASA's YO-3 provide a unique capability to help untangle the problem. The problems with using the YO-3, however, are its limited forward speed of around 110 knots and its relatively slow rate of climb as compared to most modern helicopters. Even so, it can provide very useful data in its current state. In addition, the capability of NASA's RSRA helicopter (for flight testing modified prime propulsion systems) should also be considered when defining the types of data which are needed. The Flight Research Rotor (FRR) program and the planned four bladed instrumented blade program are examples of programs which can provide valuable acoustic data. Wind tunnels which are acoustically treated are probably our prime facilities for studying and conducting research on the individual mechanisms associated with rotor noise. These facilities include model scale facilities (such as the United Technologies Research Center tunnel, the MIT tunnel and the German-Dutch tunnel) and the full scale NASA Ames 40x80 facility. The NASA Langley V/STOL tunnel can also provide some useful, but limited, data in its current untreated state. Acoustic treatment of this tunnel facility would go a long way in providing the facilities needed for this program. Serious consideration should also be paid to the possibility of updating the YO-3 to higher speeds and rates of climb (if possible).

TARGETS OF OPPORTUNITY FOR 1st PRINCIPLE ANALYSIS DEVELOPMENT & CORRELATION

- **RSRA**
- **FRR**
- **YO-3**
- **WIND TUNNELS (UTRC, V/STOL, GERMAN-DUTCH, TREATED 40x80)**

Figure 7

CURRENT INDUSTRY DESIGN/ANALYSIS APPROACH

At the start of any design, the acoustician is faced with a judgment evaluation of his analysis capability. In applying his past experience to this question he must assess the risk of applying this analysis to the design and then must define the design margins which he must apply to achieve an acceptable probability of certification. The figure below illustrates some of the factors which must be considered in this determination. Some of the subsequent figures will address these factors individually.

THE DESIGNER MUST RELY ON SEMI-EMPIRICAL ANALYSES

BASIC TASK-ASSESS RISK & DEFINE DESIGN MARGINS TO BE USED

CONSIDERATIONS

- **ACCURACY OF DESIGN METHODS**
- **ACCURACY OF TEST TECHNIQUES**
- **RISK REDUCTION TOOLS AVAILABLE**
- **KNOBS TO TURN (FALL BACK POSITIONS)**
- **CONSEQUENCES OF FAILURE**

Figure 8

ACCURACY OF COMPONENT SEMI-EMPIRICAL ANALYSIS

An earlier talk addressed the overall results of an SAE-A-21 V/STOL Noise Committee study which evaluated the accuracy of applying existing component semi-empirical analyses to several current helicopters. The results, when compared with the measured levels of these helicopters, were not very encouraging. The following figure illustrates two more results from this study which show that even the analysis of one's own aircraft did not improve the accuracy of prediction.

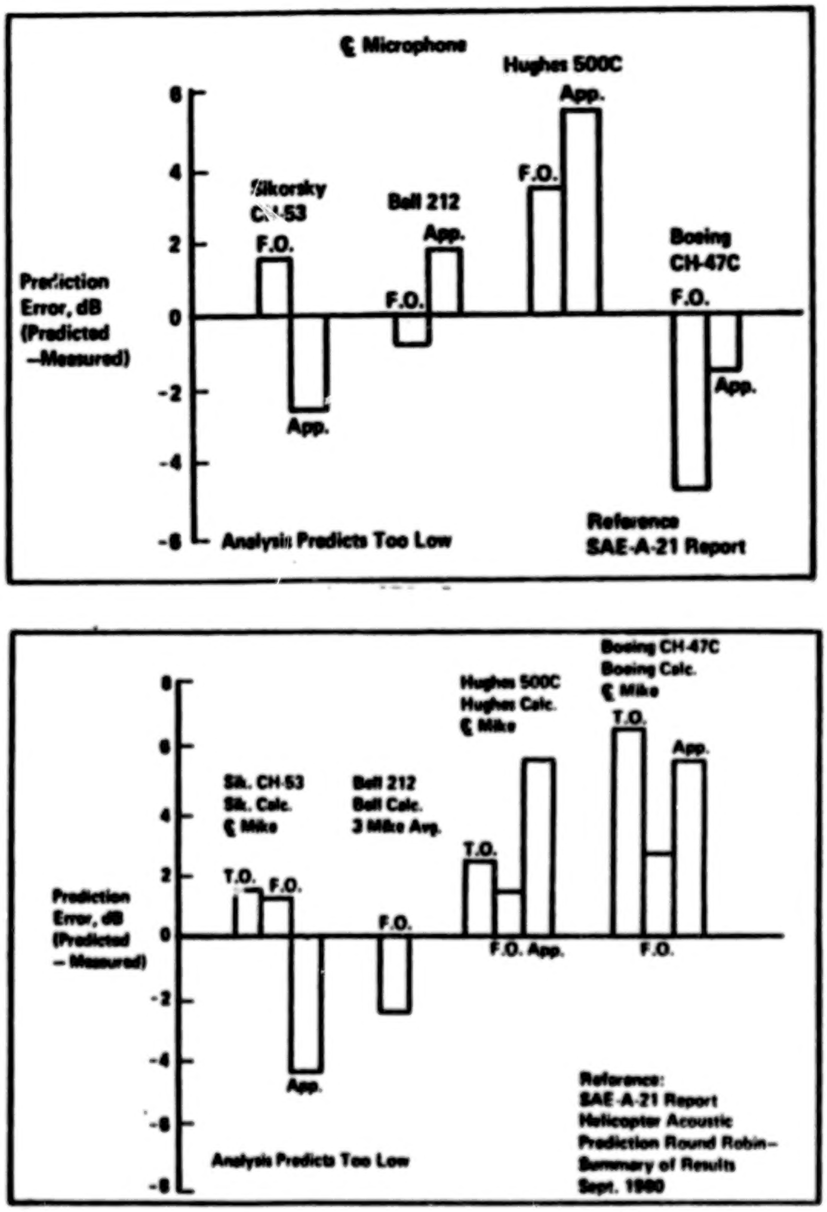


Figure 9

ACCURACY OF SYSTEM SEMI-EMPIRICAL ANALYSIS

The limited accuracy of component based semi-empirical analysis methods forced industry to explore other methods of improving reliability. One of these methods was to look at measured helicopter data and define the trends with factors shown by past experience and analysis to be primary controlling factors. Such an approach was taken by Sikorsky Aircraft. We developed a system level prediction based on disc loading, advancing tip Mach number, and rotational tip Mach number. The results of this analysis approach, when applied to available world-wide helicopter noise data, are shown in the figure below. As can be seen, there still remains a fairly wide scatter of data.

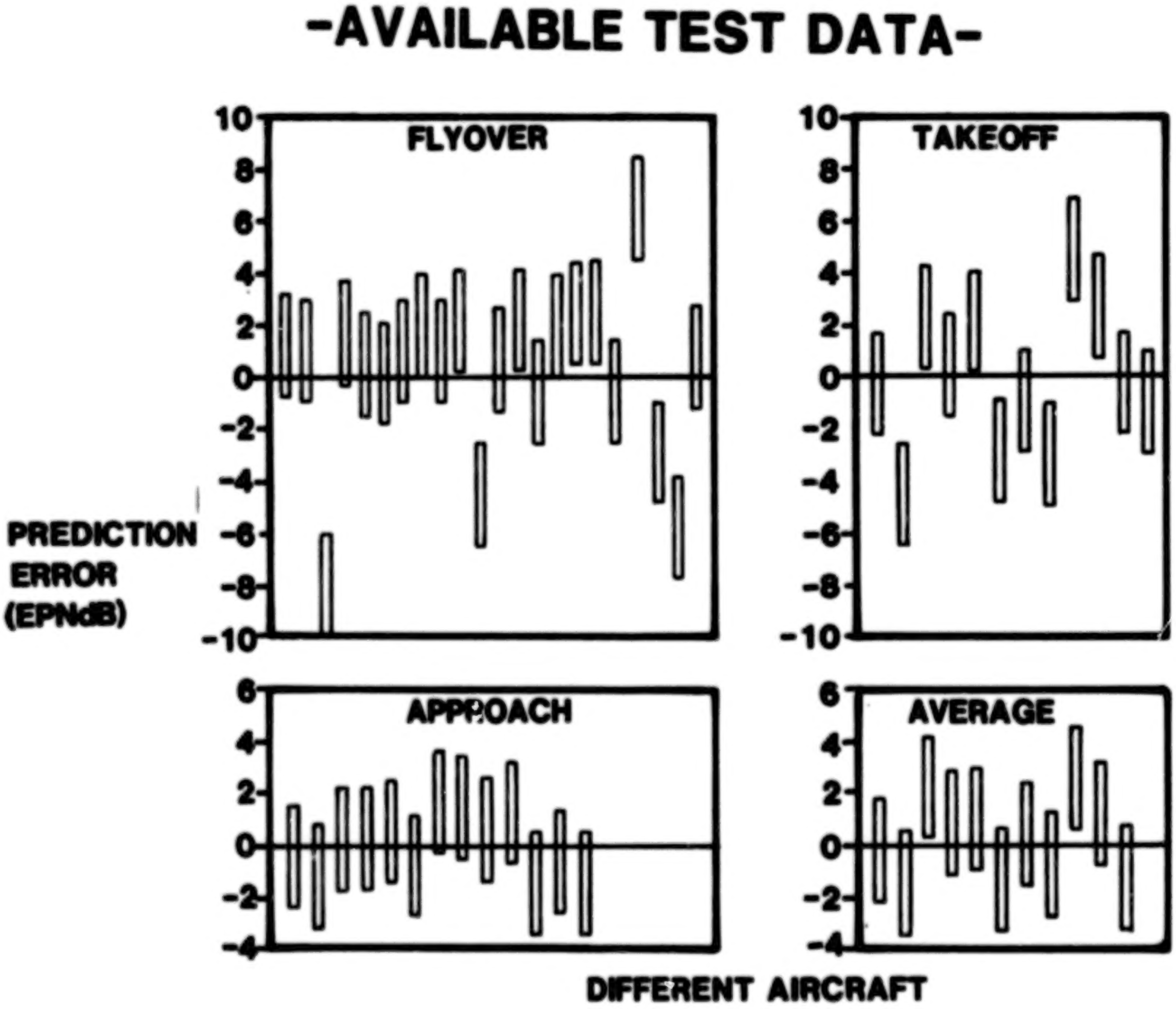


Figure 10

FURTHER SYSTEM SEMI-EMPIRICAL ANALYSIS CORRELATION

When the aforementioned analysis was applied to recent acoustic data acquired by the FAA on the Sikorsky UH-60A (BLACK HAWK) helicopter and the Sikorsky S-76 helicopter, the results show that both helicopters would have been overpredicted by from 2-5 EPNdB throughout their high speed ranges. The advanced technology incorporated into the blade design on these two helicopters is the probable reason for this difference.

COMPARISON OF SYSTEM SEMI-EMPIRICAL PREDICTIONS WITH TEST DATA ON TWO SIKORSKY HELICOPTERS

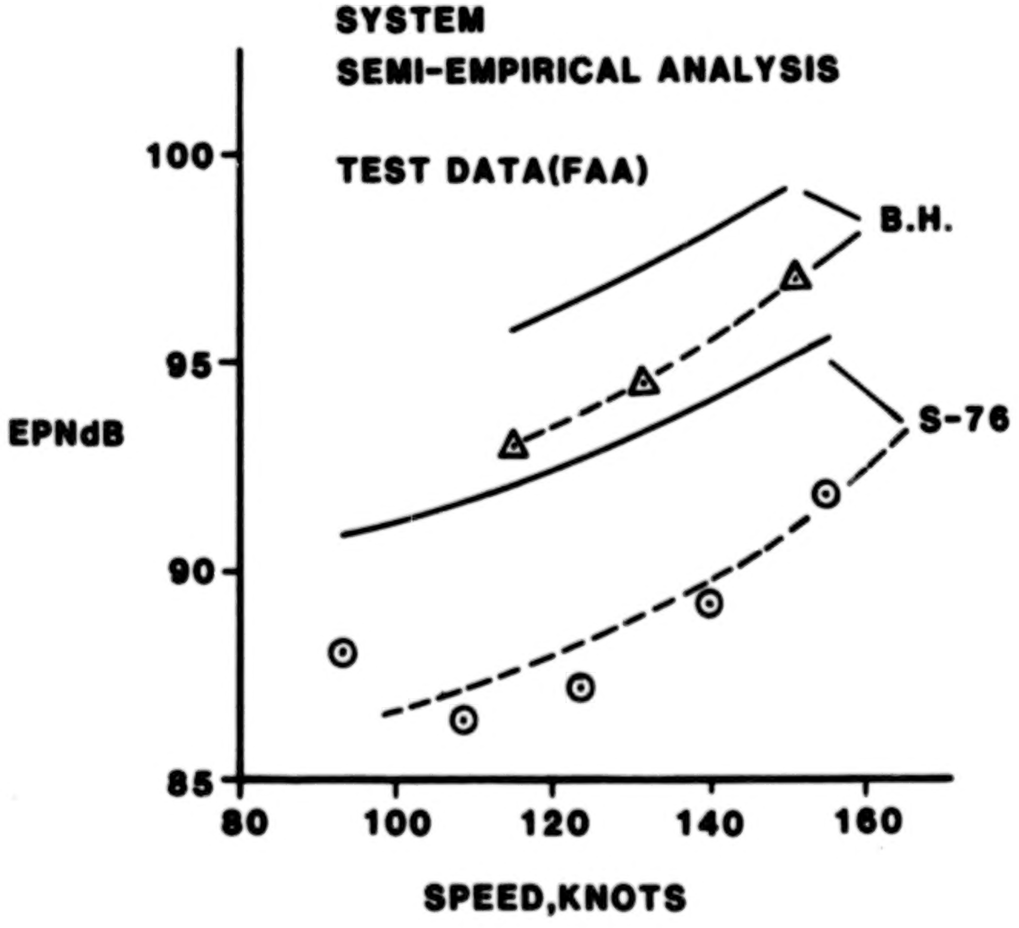
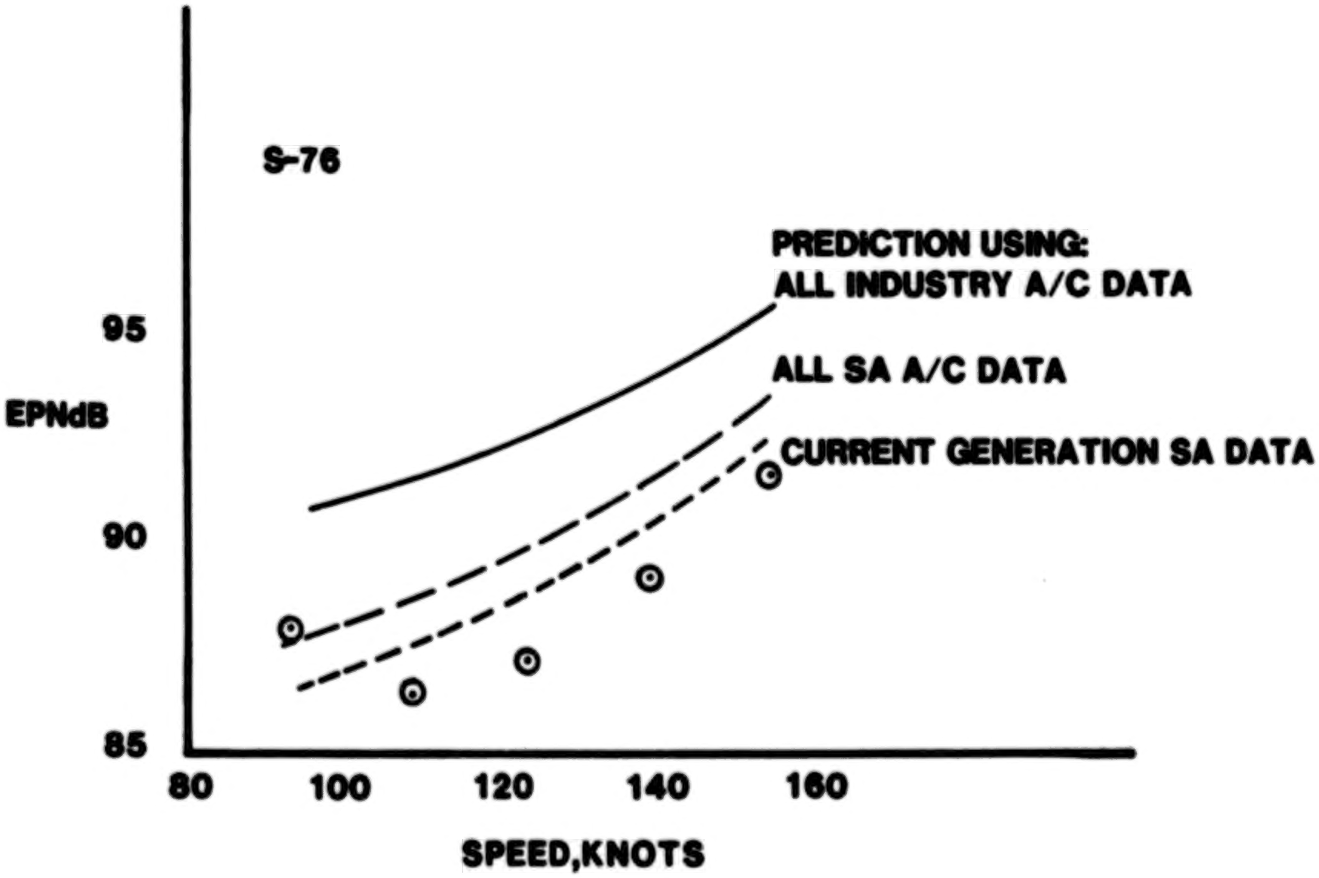


Figure 11

SYSTEM SEMI-EMPIRICAL ANALYSIS REFINEMENTS

If one considers only one's own company's products, then company design practices can be factored into the system analysis approach to further reduce prediction scatter. When this is done for Sikorsky products, a curve which is 2-3 dB below the world fleet helicopter line results. This is illustrated in the figure below. Current generation Sikorsky helicopters are still overpredicted by an average of around one dB, and the S-76 would be overpredicted by from 3 to 5 dB throughout the speed range of 90 to 155 knots.



PREDICTION USING SELECT DATA SETS

Figure 12

ACCURACY OF TEST DATA

Most data accuracy figures quoted today relate to the accuracy of data acquired by a single source on an individual aircraft over a relatively short period of time. When considering the probability of repeating past measurements, however, one must also consider test-to-test differences on similar aircraft. All of the data shown below were measured under controlled ICAO test conditions and were reduced per ICAO Annex 16 standards, but were acquired at different periods of time and presumably on different aircraft of the same model. The latest set of data on the A-109 is the most recent example of test-to-test differences which, to the author's knowledge, still remain unresolved. Test-to-test differences of up to 5 EPNdB are not uncommon.

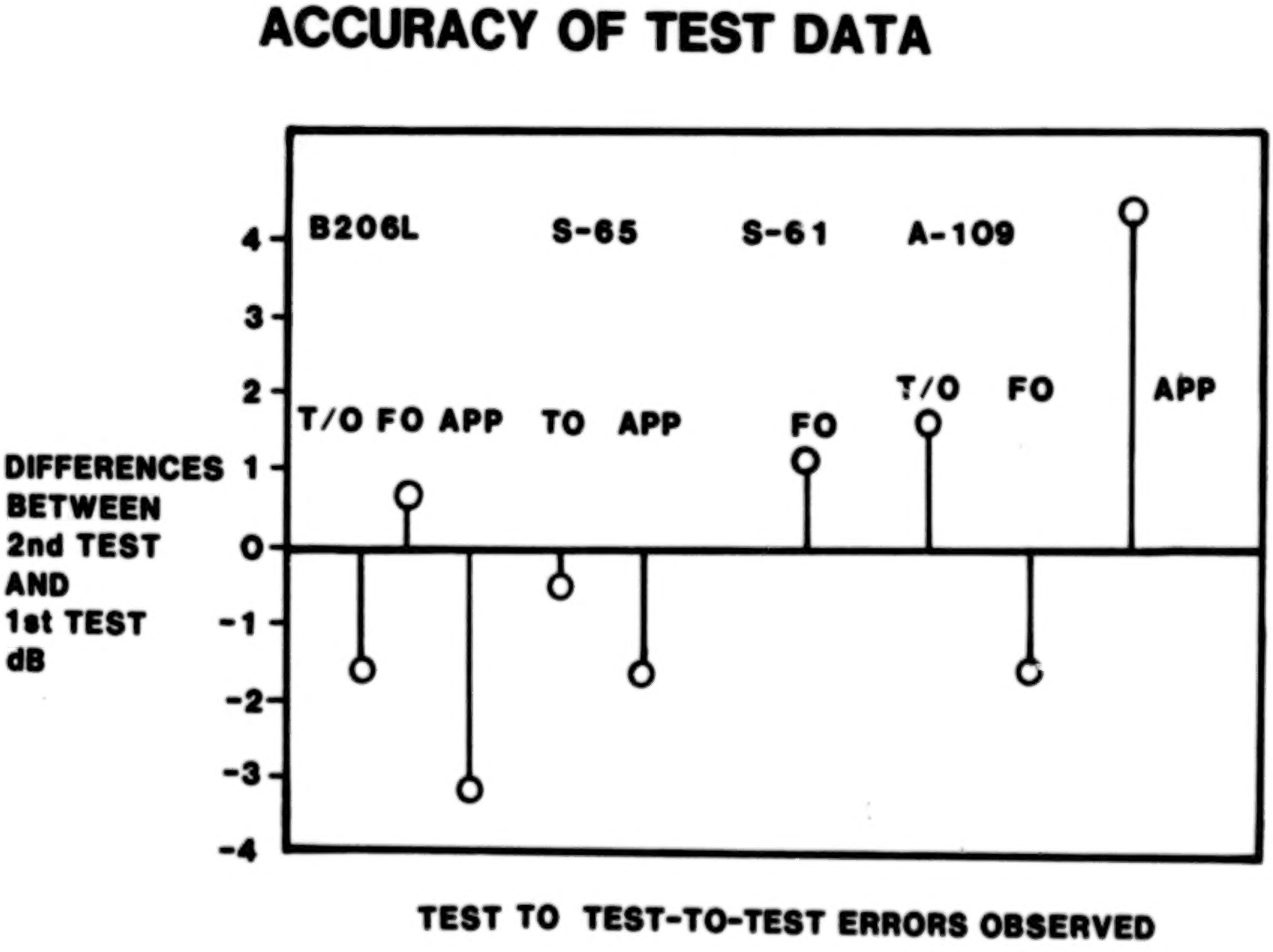


Figure 13

RISK REDUCTION ACOUSTIC WIND TUNNEL TEST FACILITIES

The industry needs an economic means to reduce the risk of design from an acoustic point of view. We must address the acoustic problem in the same manner as we have developed wind tunnels to evaluate the aerodynamics of model and full scale hardware/vehicles prior to flight. Unfortunately, U.S. facilities to accomplish this are limited. Industry has developed the technology to allow testing of around 1/6 scale model rotors and vehicles to evaluate their aerodynamic characteristics. We not only must develop this same technology for acoustics, but must also improve the acoustic characteristics of these tunnels to allow this. The figure below compares the capabilities of several wind tunnels for use in measuring S-76 helicopter model noise at a typical high speed flyover condition. Positive numbers signify usefulness, where the rotor noise signal at a distance of three rotor diameters (the approximate boundary of near field and far field noise for many sources) is higher than the tunnel noise at frequencies of importance to the civil problem. This is not to say that tunnels like NASA's V/STOL tunnel cannot be used for acoustic purposes (in particular for takeoff and approach conditions), but rather that acoustic differences at high speeds can generally only be evaluated for discrete frequency sources (not broadband) and only in the near field. U.S. wind tunnel facilities must be developed to address the helicopter's civil problem, as well as the military problem, which is one of both harmonic and broadband far field noise.

WIND TUNNELS FOR ACOUSTIC RISK REDUCTION TESTING ARE LIMITED

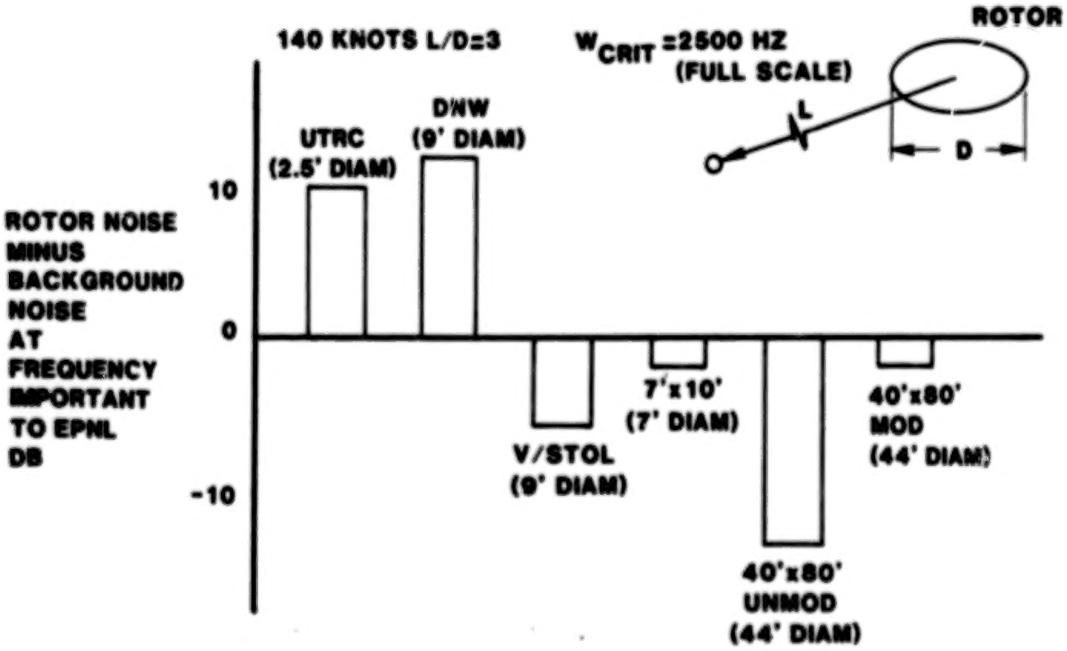


Figure 14

PROBABILITY OF SUCCESS

Theoretically, one can calculate the probability of success of a given design to achieve prescribed noise levels if the accuracies of prediction and test are known. An example of a calculation procedure for evaluating the probability of passing the proposed FAA helicopter noise rule is shown below. This relationship holds only when the seven conditions to be met are independent variables. Since the proof of this independence cannot be taken as a foregone conclusion in the case of the helicopter noise conditions to be met, analytic considerations must be made to assure that the conditions are properly handled.

PROBABILITY PROCEDURE

TO PASS THE PROPOSED FAA NOISE REGULATION ,THE FOLLOWING 7 CONDITIONS MUST BE SATISFIED:	PROBABILITY OF PASSING SUCH CONDITION
1. TAKEOFF NO GREATER THAN 2 dB OVER	P_1
2. FLYOVER NO GREATER THAN 2 dB OVER	P_2
3. APPROACH NO GREATER THAN 2 dB OVER	P_3
4. T.O.+ F.O. NO GREATER THAN 3 dB OVER	P_4
5. F.O.+APP. NO GREATER THAN 3 dB	P_5
6. T.O. + F.P. NO GREATER THAN 3 dB OVER	P_6
7. T.O.+F.P.+APP. NO GREATER THAN 0 dB OVER	P_7

PT =TOTAL PROBABILITY OF PASSING =$P_1 P_2 P_4 P_5 P_6 P_7$

Figure 15

PROBABILITY EXAMPLE

In the world-wide industry report (commonly referred to as the "CEO Report") which was submitted to the FAA in December of 1980 as regards the FAA's proposed helicopter noise rule, NPRM 79-13, an example of the probability calculation was included, which is shown below. The calculation addressed the requirement for condition independence by eliminating in subsequent calculations those populations or data ranges which resulted in failing previous conditions. The details of this procedure are somewhat complex and are described in detail in the CEO Report. Once you calculate the probability of passing for each condition, you can then calculate the overall probability of passing all of the conditions.

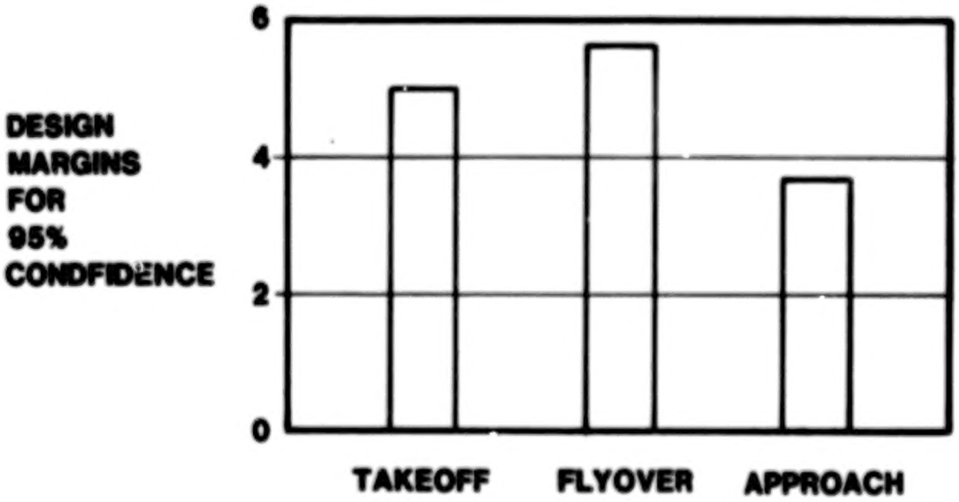
CONDITION	LIMIT +2dB	AIRCRAFT NOISE LEVEL	MARGIN	STD. DEV.	MARGIN STD.DEV.	PROBABILITY OF PASSING
1. T.O.	95.5	95.0	0.5	1.22	0.410	.659
2. F.O.	94.5	93.8	0.7	1.22	0.574	.717
3. APP	96.5	95.7	0.8	1.22	0.656	.744
4. T.O. F.O.	Δ_T=	.318				.950
5. F.O.+APP	Δ_F=	.314				.959
6. T.O.+APP	Δ_A=	.309				.964
7. T.O.+F.O. +APP						.935
			TOTAL PROBABILITY			.289

Figure 16

DESIGN MARGINS

The details of the exact mathematical procedure for determining the probability of passing regulations and defining required design margins are currently under considerable discussion by ICAO members. I will not try to defend one procedure over the other at this time. It is worthy of note, however, that the results of these other proposed procedures all seem to give similar results when one assumes similar standard deviations for measurement and prediction accuracy. The real bottom line to all of this is what do the various manufacturers around the world use as the acoustic design margins necessary to design their aircraft with confidence for the acoustic levels. A survey of the world's helicopter manufacturers was made in 1980 relative to design margins being used to develop a high confidence (95%) solution. These margins varied from around 4 to 6 EPNdB, as shown in the figure below. Since that time, U.S. manufacturers have arrived at a consensus (using a "Monte Carlo" probability analysis as a basis) of 3.5 EPNdB for each condition, if the aircraft being considered is the derivative of an acoustically well defined parent aircraft whose type charge has undergone previous acoustic evaluation, and of 5.5 EPNdB for the design of new helicopters and of derivatives which do not fulfill the aforementioned conditions.

MARGINS ACCOUNT FOR BOTH ANALYSIS AND TEST UNCERTAINTIES



RECENT U.S. INDUSTRY CONSENSUS

NEW DESIGNS-5.5 EPNdB MARGIN

DERIVATIVES -3.5 EPNdB MARGIN

Figure 17

IMPACT OF DESIGN MARGINS

The sensitivity of an aircraft design to the probability of certification, and hence to the design margin, can be seen in the figure shown below. Here, a preliminary design study was conducted on a 30,000 pound civil helicopter with an attempt to meet the FAA NPRM 79-13 proposed rule. The minimum weight solution which might have resulted in the absence of a rule had a calculated probability of passing the proposed acoustic certification rule of only around 50%. Design modifications which allowed only a 2 EPNdB design margin, and no growth margin, only achieved an 80% probability of certification, increased the gross weight by around 2% and started to impact the direct operating cost. No acceptable solution was found for a design with design margins similar to those previously discussed, as tip speed, disc loading and forward speed constraints started to rapidly increase costs and weights beyond those which were acceptable. Derivative design margins were shown to further aggravate this situation by causing a decrease of around 10% probability of certification for each additional decibel of margin added.

EFFECT OF ACOUSTIC DESIGN MARGIN REQUIREMENT ON DIRECT OPERATING COST OF POSSIBLE NEW DESIGN

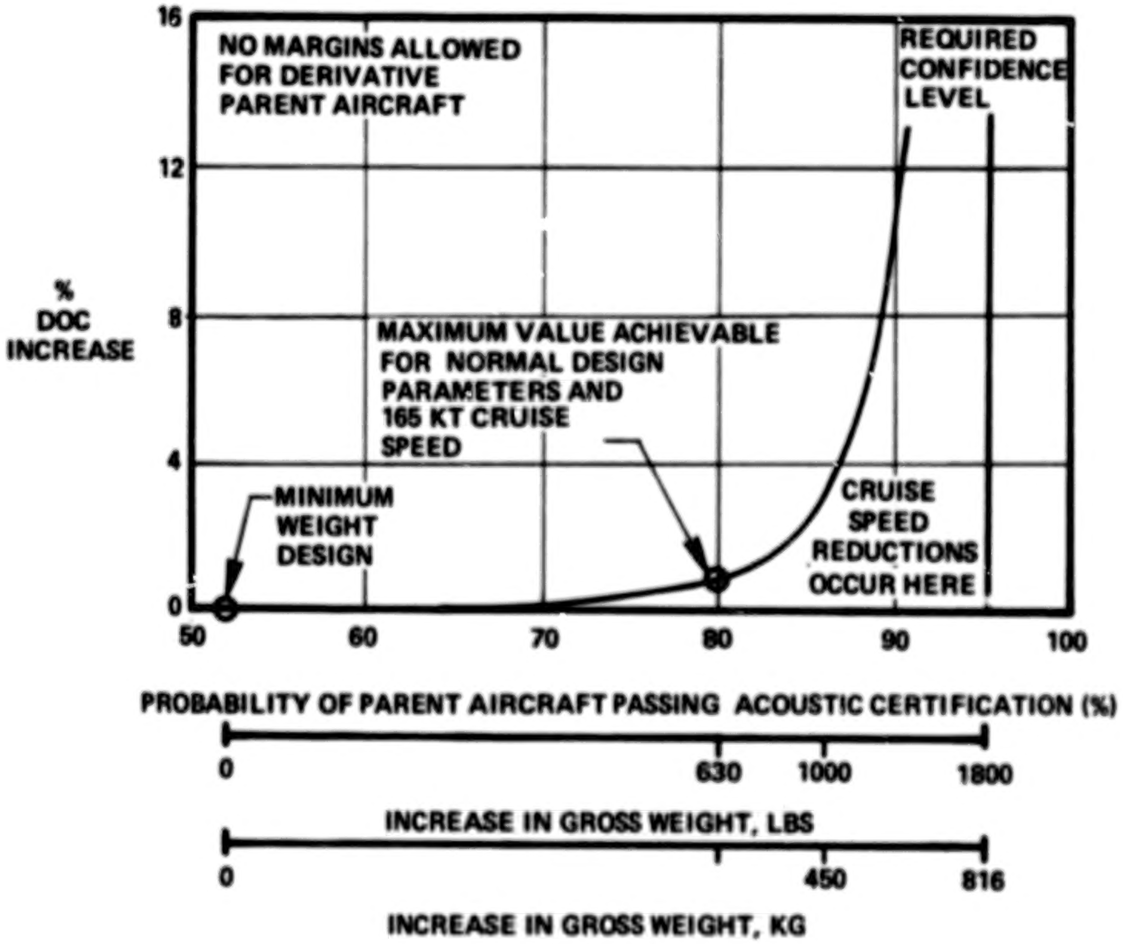


Figure 18

CONSEQUENCES OF FAILURE

The consequences of failure to achieve an acceptable design solution are fairly obvious. First, you may never go forward with your program if management is not convinced that sufficient confidence exists in your ability to meet required acoustic levels. Assuming that management proceeded with the planned program, some of the possible consequences are shown below. The performance and cost penalties associated with designing to meet the proposed FAA certification noise rule appear to be prohibitive if sufficient margins are employed which allow a high probability of success (given the inaccuracies in prediction and measurement). I hope it is obvious that a better understanding of measurement, prediction, and means of control is necessary to produce new vehicles with a high confidence not only of meeting the rule, but also of achieving a high level of community acceptance.

CONSEQUENCES OF FAILURE

- **DELAYED CERTIFICATION**
- **LOSS OF COMPETITIVE CONTRACT**
- **LOSS OF PORTION OF INCENTIVE FEE**
- **SUBSTANTIAL PERFORMANCE PENALTY**
- **(WHAT ADDS-"EXPERIENCED ACOUSTICS ENGINEER")**

Figure 19

FALL BACK POSITIONS

Once your company commits to a program, you would like to have some "knobs to turn" in the vehicle design in case you fail to meet the prescribed acoustic requirement. The problem is that available solutions are very costly, as shown in the figure below. A one EPNdB reduction would require a decrease in vehicle gross weight by from 7 to 14 percent, based on the industry trends of from 5 to 10 dB per doubling of gross weight. This decrease in gross weight, if taken from useful payload, could result in as much as a 70% payload reduction. If one was able to reduce tip speed or forward speed, reductions on the order of 8-10% to each would be required for this same one decibel reduction in the acoustic level. More cost effective fall back solutions must surely be developed.

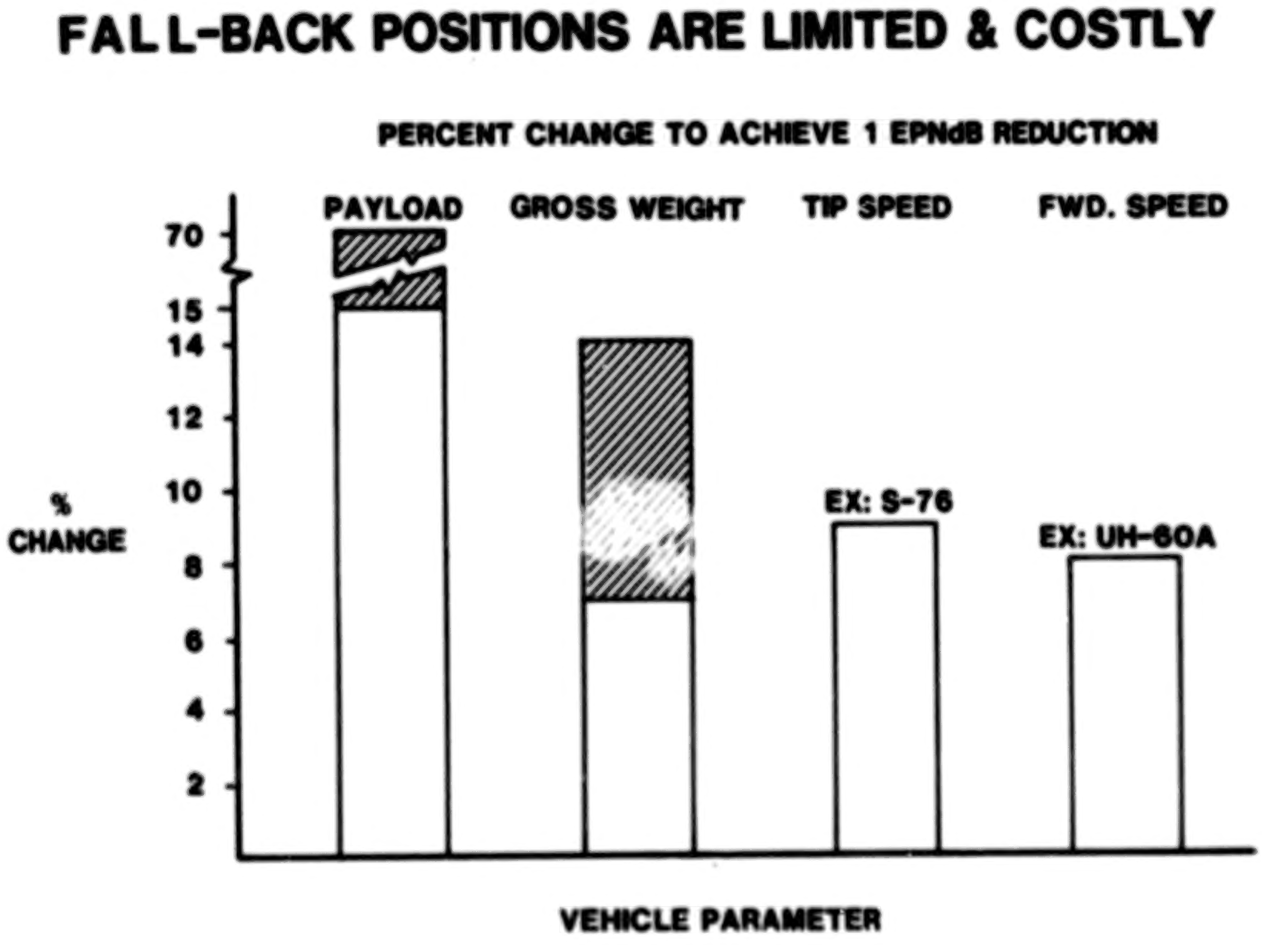


Figure 20

DESIRED METHODOLOGY

Some of the acoustic methodology requirements are delineated below. Industry must have the capability of evaluating the noise levels of preliminary design vehicles prior to the time when all of the detailed design information has been defined. This can be either empirical, semi-empirical or trend information from 1st principle system design studies (when such analysis and studies are available). Once you have settled on a design which you feel will come reasonably close to the acoustic criteria used in the preliminary design cycle, you define the design details necessary for a 1st principle analysis approach. In addition, the much more simple problem of predicting the acoustic effect of minor vehicle modifications must also be addressed. This can normally be accomplished through empirical trend measurements on the vehicle in question (or at least one that is similar). Today's state of technology requires that you commit to design with a lesser level of acoustic confidence than you need for a high confidence solution. This approach requires that you have means for reducing your risk during the program. Acoustic wind tunnel model testing and cost effective "knobs to turn" are both needed to address this need.

DESIRED METHODOLOGY

- NEW "PAPER" AIRCRAFT PREDICTION METHODS
  - PRELIMINARY DESIGN
  - DETAIL DESIGN
- DERIVATIVE AIRCRAFT TREND PREDICTION METHODS
- CAPABILITY TO CONFIRM DESIGN THROUGH MODEL TESTS
- HIGH BYPASS RATIO EQUIVALENT SOLUTION TO HELICOPTER NOISE
- FALL BACK "KNOBS TO TURN"

Figure 21

CONCLUSIONS

Helicopter acoustics today remains more of an art than a science, and manufacturers must rely heavily on past experience in the designs of new vehicles constrained by acoustic standards, regulations, contractual requirements or guidelines. There has been a substantial research and development effort identified to improve this situation. One of the problems is, however, that there are so many elements of the problem that early efforts must be made to better define and prioritize the problem. This can be done through careful measurements and analysis of current technology helicopters and hardware through the use of vehicles and facilities such as the RSRA and the YO-3. The problem appears also to be long term in its solution. The development of regulations and standards must be mindful of this status in their formulation in order to assure that they are economically reasonable and technologically practical.

CONCLUSIONS

- **HELICOPTER ACOUSTIC PREDICTIONS & CONTROL NOT YET A SCIENCE**
- **SUBSTANTIAL R&D EFFORT APPEARS NECESSARY**
- **INITIAL EFFORT MUST BETTER DEFINE THE PROBLEM TO PRIORITIZE & DIRECT SUBSEQUENT RESEARCH EFFORTS**
- **REGULATIONS MUST BE MINDFUL OF TECHNOLOGY STATUS**

Figure 22

SPECIAL SESSION

PSYCHOACOUSTICS AND HELICOPTER PROPULSION NOISE

Session Moderator: R. J. Huston
NASA Langley Research Center

N84 29675

PSYCHOACOUSTIC CONSIDERATIONS FOR HELICOPTER NOISE QUANTIFICATION: METRICS, HUMAN RESPONSE, AND CRITERIA

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INTRODUCTION

The quantification of helicopter noise for human response purposes has been the subject of research for over 21 years (ref. 1). Although the quantification of the "noisiness" or annoyance potential of helicopter noise has not received as much research attention as has the quantification of jet airplane noise, a number of studies (refs. 2 and 3 for example) have indicated that helicopter noisiness is not as reliably predicted by most noise metrics as is the noisiness of jet airplanes. The most probable reason is that the character of helicopter noise is more diverse than that of jet airplanes. Each of the primary noise sources--main rotor, tail rotor and engine--produces distinctive noises which are quite variable between different models and under different operating conditions. Consequently, a metric for accurately quantifying helicopter noise must be able to account for a wide range of spectral and temporal characteristics.

The purpose of this paper is to examine several aspects of helicopter noise quantification from the standpoint of psychoacoustics. The topics to be considered are indicated in figure 1. First, noise metrics in common use to describe far-field aircraft noise and the noise characteristics which they consider are discussed. Second, some findings of recent psychoacoustic research related specifically to helicopter noise quantification are presented. Third, criteria for the accuracy of noise metrics to quantify helicopter noise are discussed. Finally, the prospects for improved metrics and research needed to develop and validate improved metrics or existing metrics are discussed.

TOPICS CONSIDERED

- NOISE METRICS AND PHYSICAL CHARACTERISTICS
- RECENT RESEARCH FINDINGS
- ACCURACY CRITERIA FOR NOISE METRICS
- NEEDED QUANTIFICATION RESEARCH

Figure 1

OBJECTIVES OF AIRCRAFT NOISE QUANTIFICATION

Three objectives of aircraft noise quantification which can ultimately lead to reduced aircraft community noise impact are identified in figure 2. These objectives are based on three different uses and users of noise metrics. The first objective is to promote effective noise reduction by accurately quantifying human response to noise source differences. Through the use of appropriate metrics and design trade-offs, noise control engineers can identify and reduce the most offensive components of complex noise sources. Thus, the most cost effective noise reduction can be effected. The second objective of aircraft noise quantification is to provide equitable noise certification. If the certification noise levels for different aircraft within a class accurately reflect human response to the noise produced by those aircraft in communities, the certifying authority is assured that community noise impact is being limited. Equally important, manufacturers of specific aircraft are not being unduely penalized nor rewarded by a flaw in the certification metric. The final objective of aircraft noise quantification is to provide predictable response to all types of aircraft. By accurately quantifying response to different classes of aircraft, a noise metric can be used to predict total aircraft noise impact in communities where mixed fleets exist or where different classes are to be introduced. Thus, specific classes of aircraft will not be unduely restricted and impact reduction through operational alternatives such as runway use and ground track optimization can be effected.

- **EFFECTIVE NOISE REDUCTION--QUANTIFY SOURCE DIFFERENCES**

- **EQUITABLE NOISE CERTIFICATION--QUANTIFY WITHIN CLASS DIFFERENCES**

- **PREDICTABLE RESPONSE TO ALL TYPES--QUANTIFY BETWEEN CLASS DIFFERENCES**

Figure 2

NOISE CHARACTERISTICS CONSIDERED FOR QUANTIFICATION

The noise characteristics generally considered to quantify the noisiness of all aircraft are indicated in figure 3. Most metrics consider the spectral characteristics of the measured sound. This consideration can be a single frequency weighting function to approximate human hearing frequency sensitivity, which can be incorporated in sound level meters. It also can be a complicated calculation procedure which approximates the frequency, frequency and level interaction and masking sensitivity of the human auditory system. Some metrics also consider the possibility of increased noisiness due to discrete tonal components. Other metrics consider temporal characteristics of the measured sound by integrating the total energy in a noise event. Thus, some metrics provide consideration of both duration and fluctuations in level of flyover noises in addition to the spectral content. Some metrics, sometimes referred to as noise indexes, consider other aspects of noise impact more related to actual operations. These metrics commonly consider the number of operations, generally in an energy sense so that a doubling of number is equivalent to a 3 dB increase in level. Some metrics also consider the time of day of occurrence of noise events.

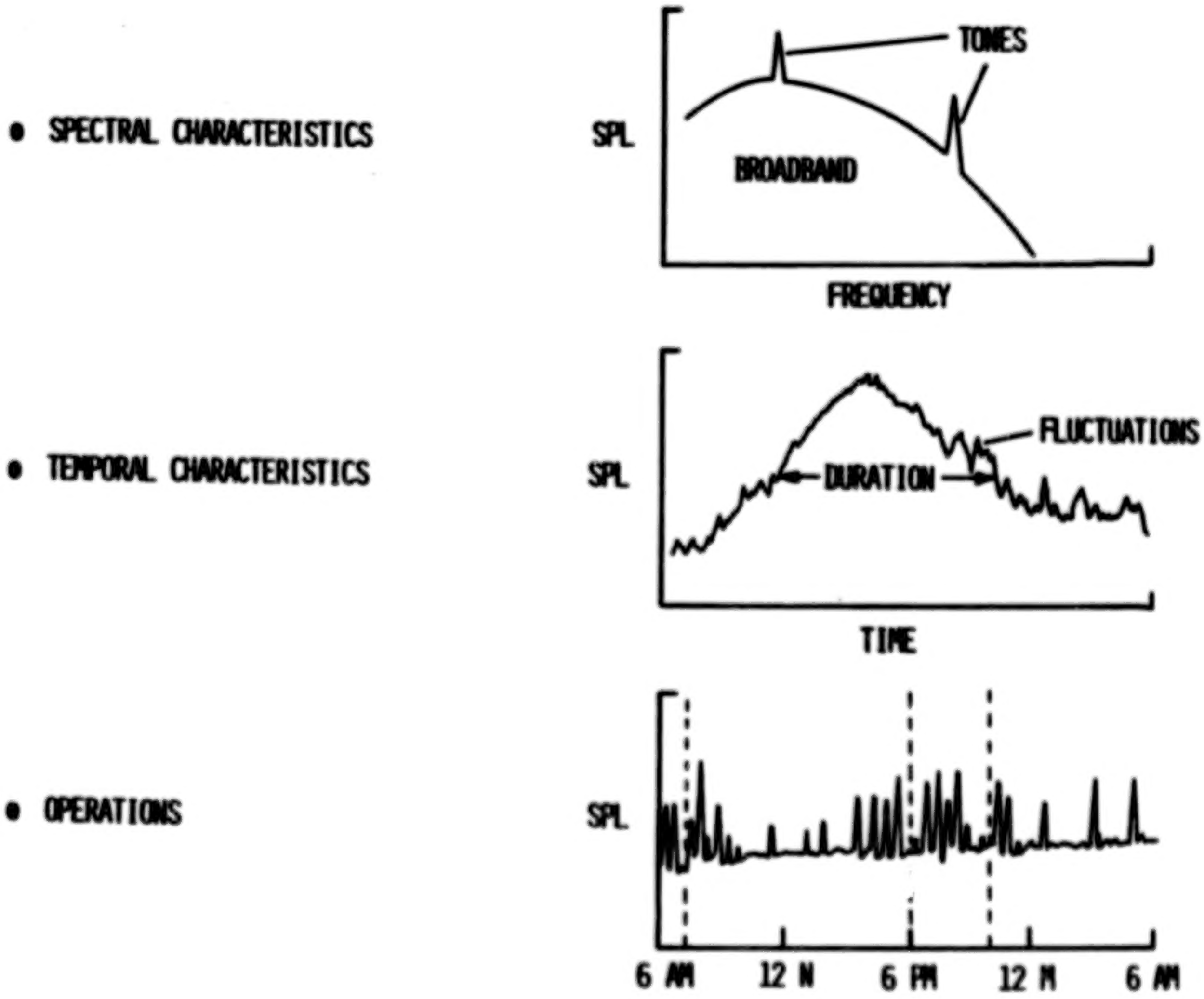

Figure 3

SPECIAL NOISE CHARACTERISTICS OF HELICOPTERS

The diverse nature of helicopter noise provides a somewhat greater challenge for metrics to be able to adequately quantify the annoyance potential. Some of the special characteristics which need to be considered for helicopter noise are indicated in figure 4. First, helicopter noise results from a number of different sources each of which is distinctively perceptible by people. These sources can compete by being the subjectively dominant source during various operating conditions for a given helicopter. The three major sources are the main rotor, tail rotor, and engine. One of the most easily identifiable helicopter noises is commonly called blade-slap. Two different forms of blade-slap noise can be found in some helicopters and/or during certain operations. One type results from high rotor tip speed, commonly called thickness noise; another results from blades intercepting vortices of other blades. The helicopter tail rotor is also the source of easily identifiable noise. This noise consists of harmonically related tonal components and can be characterized as sounding like an almost melodious pure tone or a rasping buzz depending on the harmonic structure. Helicopter noise also contains varying amounts of broadband noise from engines and rotors. In some instances the broadband noise produces swishing sounds because of directivity and rotor rotation effects.

In addition to the unusual spectral characteristics described, helicopter noise can also exhibit unusual durations or time histories. The directivity of main rotor thickness noise and tail rotor noise results in greater noise being radiated in the forward direction during forward flight. As a result, the increase in noise level with lessening distance can be very gradual as a helicopter approaches. After passing overhead, the noise level decreases rapidly. The strong directivity patterns and distinctive nature of these noises can also result in increased detectability at great distances.

- **COMPETING NOISE SOURCES**
- **THICKNESS NOISE BLADE-SLAP**
- **VORTEX INTERACTION BLADE-SLAP**
- **TAIL ROTOR TONES**
- **BROADBAND NOISE**
- **UNUSUAL TIME HISTORIES**

Figure 4

NOISE CHARACTERISTICS CONSIDERED BY NOISE METRICS

As indicated in figure 3, different noise metrics of varying complexity consider different noise characteristics. The specific characteristics considered by the most common noise metrics are indicated in figure 5. The weighted metrics such as L_A, L_D, and others take into account only the general effects of spectral characteristics. The computed metrics, loudness level (LL), perceived noise level (PNL) and perceived level (PL) add spectra-level interaction and masking effects. Effective perceived noise level (EPNL) adds both duration and pure tone effects to the previous effects whereas sound exposure level (SEL or sometimes called L_{AX}) adds only duration effects to the simple spectral effects. The community noise indexes including equivalent continuous sound level (L_{EQ}) account for a number of events. The day-night average sound level (L_{DN}), community noise equivalent level (CNEL) and noise exposure forecast (NEF) also provide corrections for different time-of-day periods. Since NEF is based on the single event measure EPNL, it considers the effects of tones, spectra-level interaction and masking in addition to number of events and time-of-day.

A question of major concern is whether the most sophisticated metrics, EPNL for single events and NEF for multiple events, adequately quantify helicopter noise or aircraft noise in general. This question will be addressed in the next few figures by presenting some recent subjective research results.

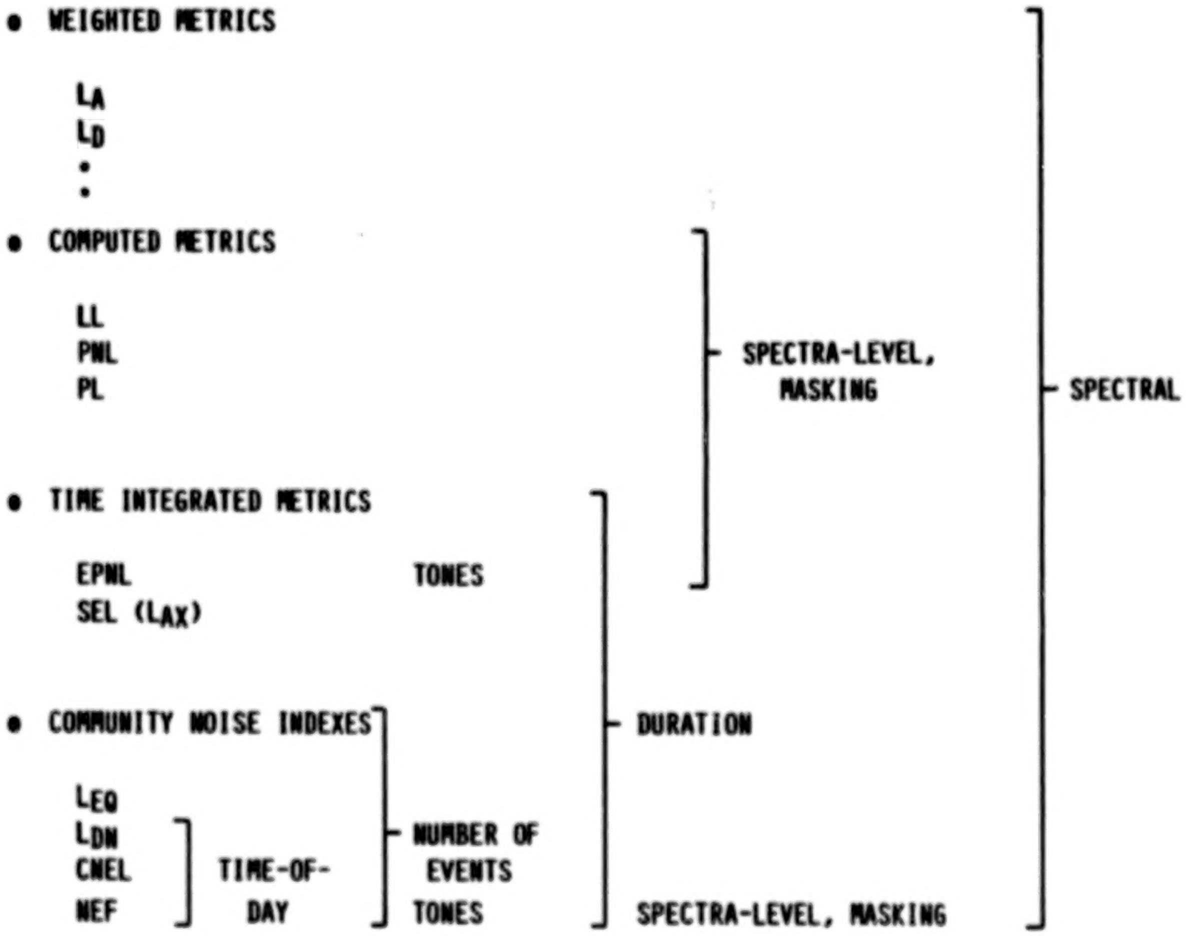
• WEIGHTED METRICS
L_A
L_D
•
•
• COMPUTED METRICS
LL
PNL
PL
SPECTRA-LEVEL, MASKING
SPECTRAL
• TIME INTEGRATED METRICS
EPNL
TONES
SEL (L_{AX})
• COMMUNITY NOISE INDEXES
DURATION
L_{EQ}
L_{DN}
CNEL
NEF
TIME-OF-DAY
NUMBER OF EVENTS
TONES
SPECTRA-LEVEL, MASKING

Figure 5

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FIELD STUDY OF NOISE SOURCE DIFFERENCES

The question of whether blade-slap noise is adequately accounted for by EPNL, the proposed helicopter noise certification measure, was investigated in a study at Wallops Flight Center (ref. 4). In the study, subjects located indoors and outdoors judged the annoyance of sounds of a propeller airplane and two helicopters flying prescribed flight paths. One helicopter produced significant and varying levels of blade-slap, but the other did not. The mean annoyance judgments of the outdoor subjects for the two helicopters are shown in figure 6 as related to the measured noise levels in EPNL. These results indicated that the helicopter without blade-slap was actually more annoying at a given EPNL than was the helicopter with blade-slap. These results were subsequently substantiated in laboratory tests. One obvious difference in the noise produced by the two helicopters was that the noise of the nonslapping helicopter was dominated by tail rotor noise and the noise of the slapping helicopter by main rotor noise. The implication was that EPNL was not effectively accounting for all of the different characteristics of the helicopter noises which resulted most probably from differences in the two sources of noise, main and tail rotors.

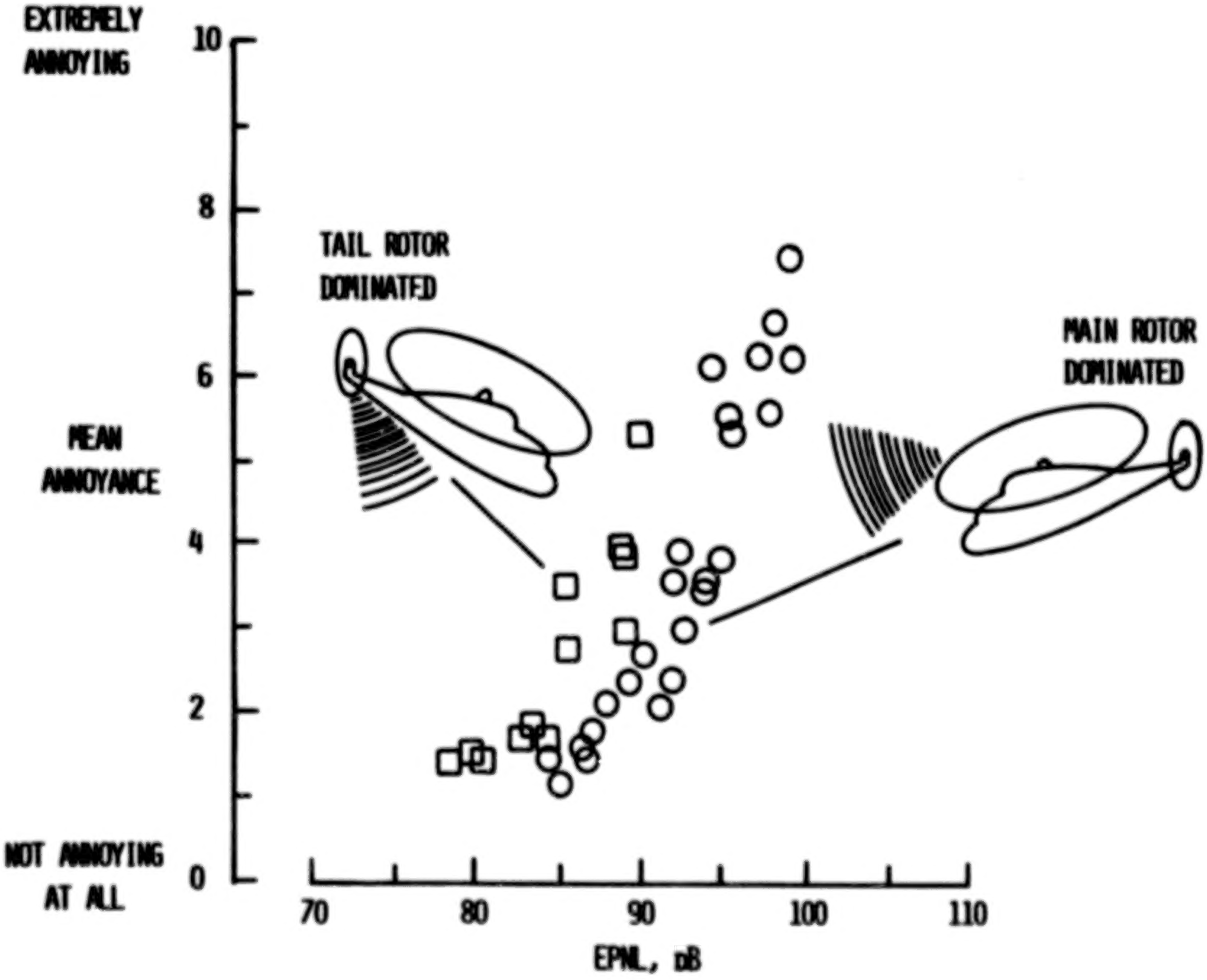

Figure 6

LABORATORY STUDY OF NOISE SOURCE DIFFERENCES

To examine the possible differences between the annoyance of the two types of rotor noise more thoroughly, a laboratory study (ref. 5) was conducted using closely controlled simulated rotor noise. Results of this study are shown in figure 7. In the study computer simulations of predicted thickness noise of a typical blade section with repetition rates from 10 Hz to 115 Hz and with tip speeds of Mach 0.63 to 0.91 were judged by subjects in an anechoic chamber. The tip speed variable controlled the sharpness or impulsiveness of the noises and, therefore, the higher frequency harmonic content. Averaged across tip speeds, the error in noisiness prediction (subjective noisiness level minus measured noise level) increased systematically with repetition rate. This increase was similar for L_A and PNL. However averaged across repetition rate, error in noisiness prediction was found to increase with blade tip speed much more for PNL than for L_A.

These results indicate that if all other factors were equal or constant, tail rotor noise would be more annoying than main rotor thickness noise because of the repetition rate effect. The results also indicate that L_A was able to compensate for the impulsive character of the noise better than PNL. Since the major difference between L_A and PNL for stationary noises is the frequency weighting, this indicated that some optimized frequency weighting could possibly offer improved annoyance prediction ability.

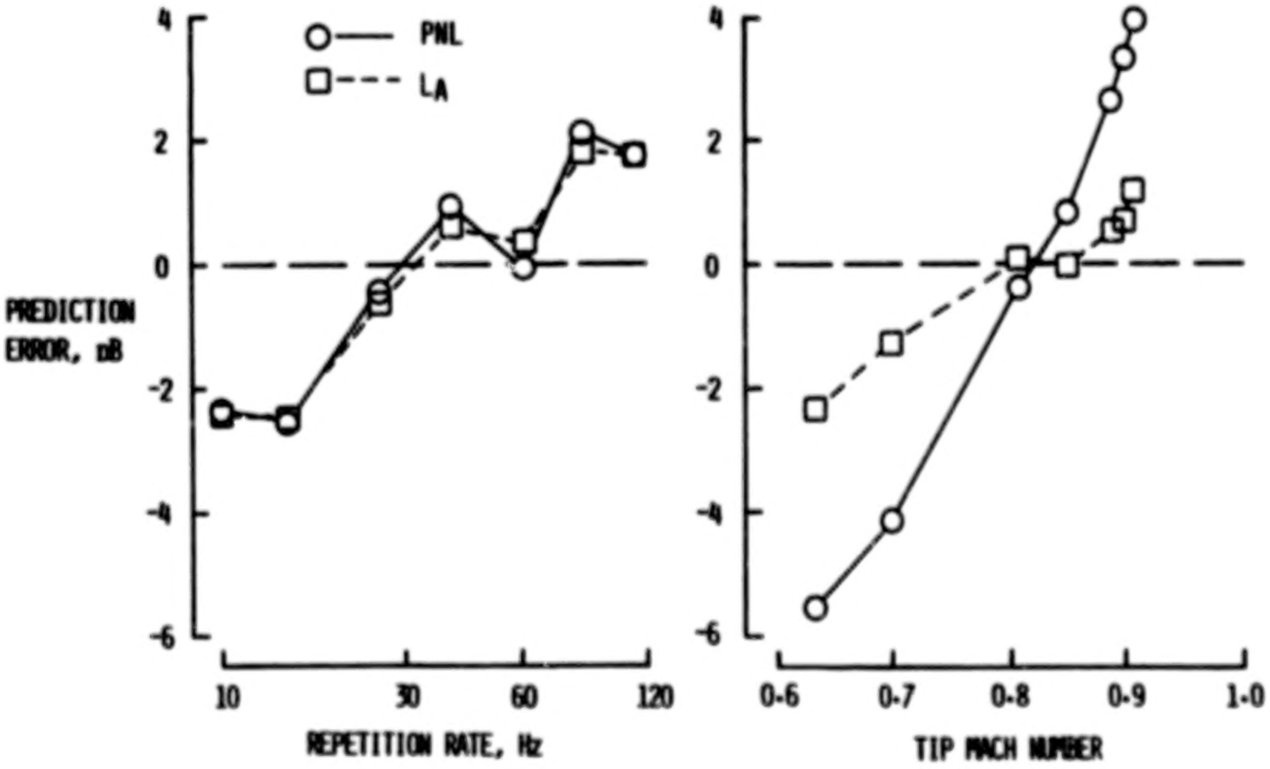

Figure 7

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WITHIN CLASS DIFFERENCES

Typical examples are illustrated in figure 8 of differences in annoyance between different helicopter types which are not accounted for by present metrics. These results were obtained in recent laboratory tests in which the noisiness of a large number of flyover noises of helicopter and jet aircraft were judged. The tests were conducted by Prof. John Ollerhead at the University of Loughborough, England on contract to NASA Langley Research Center and will be reported in a contractor report in the near future. The figure indicates the mean error in noisiness prediction for five helicopter types for EPNL and SEL. Positive errors indicate that the helicopter type was noisier than predicted by the metrics. For these particular types the results are generally consistent for the two metrics although the spread of 4-5 dB error for each metric is considered large. The data presented in this figure are based on judgments of 36 subjects on each of 4 to 6 different recordings of each helicopter type and are thereby considered highly reliable. In the tests the EPNL metric was found to be as good as any other metric; however, the variability for the different helicopter types was much larger than was found for CTOL aircraft.

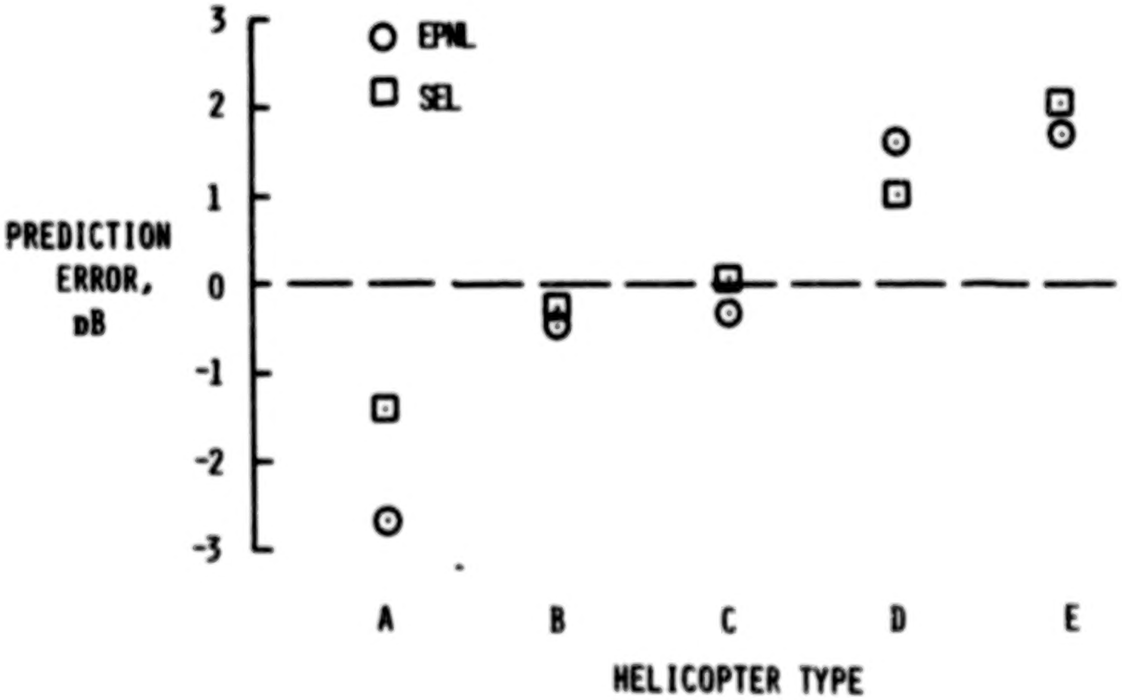

Figure 8

BETWEEN CLASS DIFFERENCES

Since a large number of both CTOL aircraft and helicopter noises were judged in the tests described in the previous slide, the tests also provide some information concerning differences between these two classes of aircraft. Figure 9 presents a comparison between the subjective judgments, converted to a decibel scale, and measured EPNL for each recorded noise. Although there is scatter about both regression lines, there is more scatter for the helicopters (standard deviations of 1.6 dB and 1.0 dB for helicopters and CTOL respectively). The primary interest in this figure, however, is the 2.0 dB average difference between helicopters and CTOL aircraft. For a given EPNL the helicopters in general were less noisy than the CTOL aircraft (predominantly jet aircraft). While this fact has very little relevance for noise certification, information of this type is important from the standpoint of community noise impact prediction and for local noise ordinances or limits. It should be emphasized that the subjective tests which provided these data were limited to noisiness of single flyover events and the question of how subjects would respond to multiple events is open to question and should be addressed in future research.

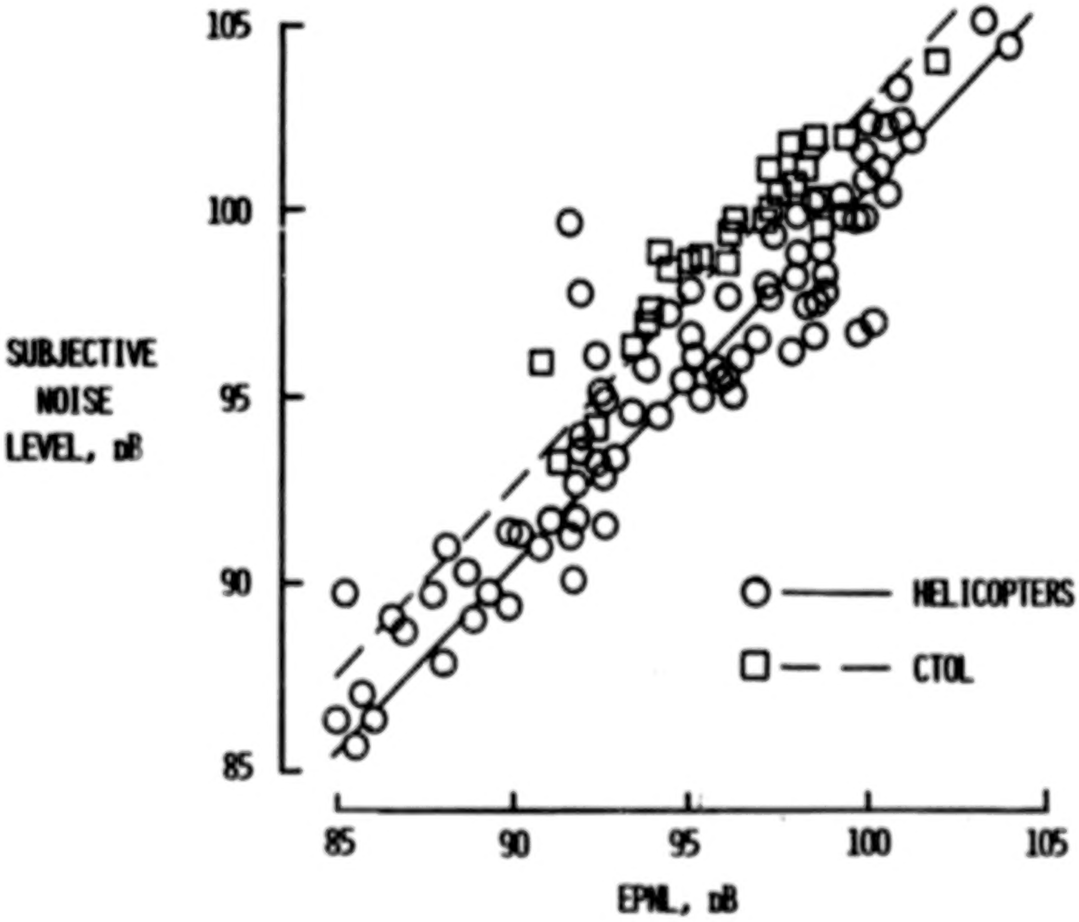

Figure 9

ACCURACY CRITERIA FOR NOISE METRICS

The basic criteria for the accuracy of noise metrics must be based, to a large extent, on the variability in people. Over the past 20 years a large volume of psychoacoustic data have been generated. Although there have been differences between test methodologies, analysis techniques, and interpretations of results to particular noises, the results concerning human variability are fairly consistent. Because of large differences between individuals, people individually make relatively poor absolute judges of annoyance or noisiness for specific sounds. However, individually or in groups they make very consistent judges of relative annoyance or noisiness between sounds. An individual will generally repeat judgments on a given sound with a standard deviation (σ) less than 1 dB. The standard deviation for discrimination of differences between sounds for a group of 25 to 50 subjects is usually less than 0.5 dB. Therefore, judged differences in sounds of anything greater than about 1 dB is usually highly significant. Based on these magnitudes of variability in people to judge sounds, reasonable criteria for the accuracy of noise metrics to quantify noisiness or annoyance potential of sounds can be established. Some guidelines are presented in figure 10.

For differences between noise sources within a complex aircraft sound, the metric should be able to predict response with a standard deviation of about 1 dB. Individuals would generally discriminate such differences. For differences within a given class of aircraft the metric should predict response with a standard deviation also of about 1 dB. Groups of people would discriminate such differences. For differences between classes of aircraft the standard deviation for predictions should be on the order of 2 dB or less. Since subject groups could be somewhat more sensitive to different classes because of past experiences or connotations of the sounds, a somewhat less reliability in prediction may well have to be expected and tolerated.

- **NOISE SOURCE DIFFERENCES $\sigma = 1$ dB**

  (BASED ON INTRASUBJECT VARIABILITY)

- **WITHIN CLASS DIFFERENCES $\sigma = 1$ dB**

  (BASED ON INTERSUBJECT VARIABILITY AND GROUP RESPONSE)

- **BETWEEN CLASS DIFFERENCES $\sigma = 2$ dB**

  (BASED ON INTERSUBJECT VARIABILITY AND GROUP RESPONSE)

Figure 10

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PREDICTION ABILITY OF PRESENT METRICS FOR HELICOPTERS

By providing a comparison of a large number of helicopter types and operations, the research conducted by Prof. Ollerhead has also provided up to date data on how well present metrics predict the noisiness of helicopters as a class. A summary of results for a number of metrics is provided in figure 11. This figure presents the standard deviation in the error between subjective response and measured noise based on both peak noise levels and duration corrected noise levels. Results are shown for three metrics with simple A, D, and E weightings and three computed metrics: PNL, PNL with tone corrections (PNLT), and PNL with an ISO proposed blade-slap correction ($PNLT_I$). As indicated, PNLT with a duration correction, in other words standard EPNL, was found to provide the least standard deviation in error. Based on these data and the metrics tested, EPNL would appear to be the most appropriate measure for predicting noisiness for helicopters at the present time. The impulse or blade-slap correction did not improve the prediction ability of the PNLT or EPNL metric.

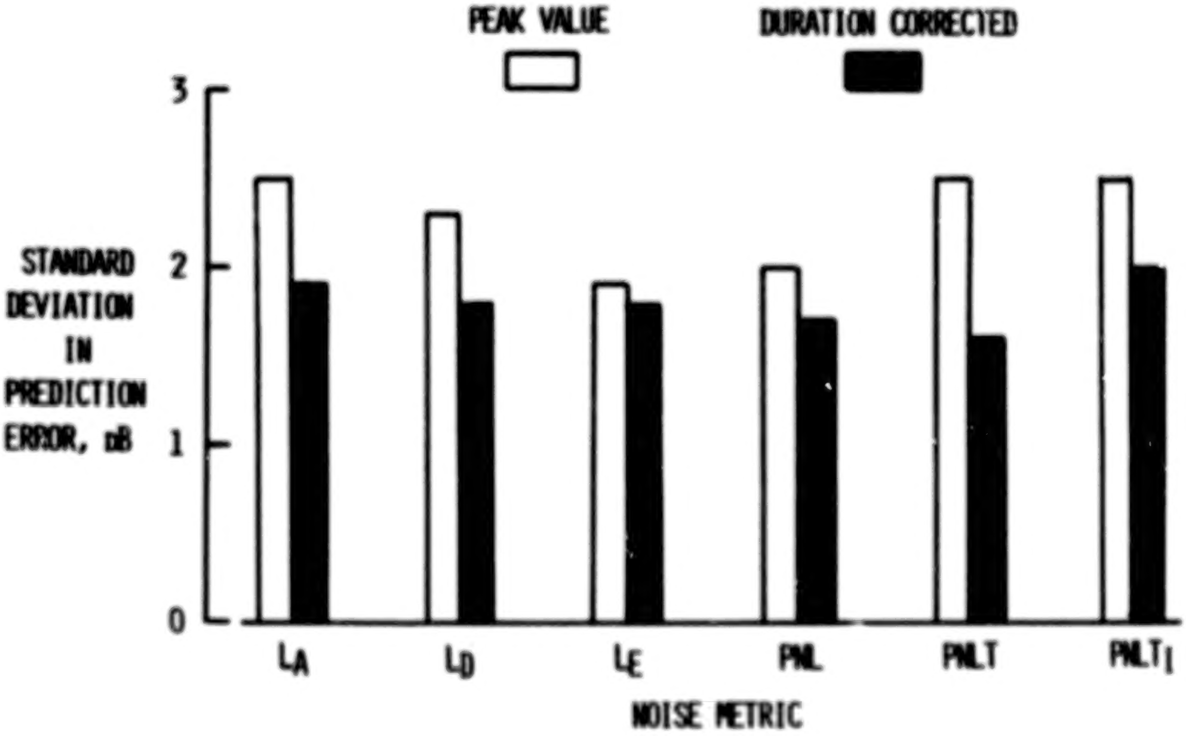

Figure 11

PROSPECTS FOR IMPROVED METRICS

Based on the results of the most recent study the EPNL metric may be the best present day metric for quantifying helicopter noise. However, based on the criteria presented in figure 10 and the results of reference 5, some of which are presented in figure 7, EPNL could be improved. Through a thorough acoustic and subjective analysis of the sounds presented in the two studies, frequency weightings which improved the noisiness prediction ability were identified. Two such weightings are presented in figure 12 and are compared with the standard A and D weightings. Based on the repetition rate and impulsiveness study of reference 5 a low frequency weighting was developed as indicated by the solid line below 1000 Hz. Based on the study conducted by Prof. Ollerhead a modified weighting above 1000 Hz was indicated. While neither of these weightings may be optimum, particularly at all noise levels, they do indicate the potential for developing improved metrics.

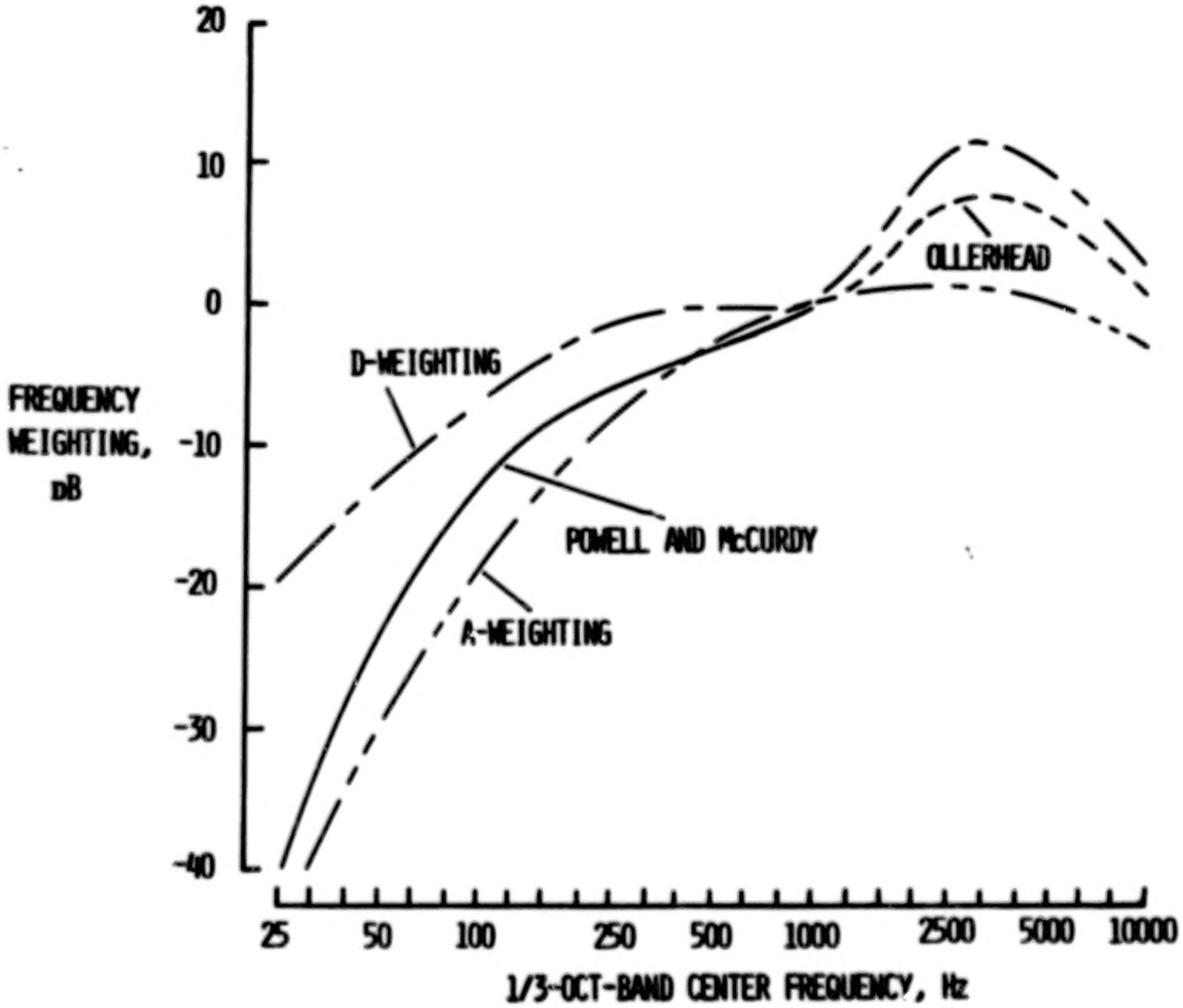

Figure 12

NEEDED QUANTIFICATION RESEARCH

Based on some of the results of recent research and a lack of research in certain areas, a number of future research topics for helicopter noise quantification can be identified and are indicated in figure 13. For single event noise metrics, recent studies have indicated the need for research in frequency weightings, particularly across the broad range of frequencies and to investigate possible interaction effects with noise level. The general lack of research as to how people subjectively integrate the effects of multiple noise sources, such as main rotor, tail rotor, and engine noise, into a single impression of noisiness or annoyance indicates a great need for future study.

For reliable multiple event noise metrics, three major areas of future research needs can be identified. One area is an investigation into a possible interaction of duration and ambient or background noise level. This is because the potential exists for enhanced detectability of the long duration sounds for helicopters with dominant blade-slap or tail rotor noise. The effects of number of helicopter operations is also an area for study. Little information exists as to whether an energy based number correction is suitable for helicopter noise community impact assessment. Finally, the need exists to compare multiple event helicopter annoyance to that of jet and propeller aircraft annoyance so that impact can be accurately assessed in communities where the different fleets coexist.

SINGLE EVENT

- **FREQUENCY WEIGHTING**
  - **BROAD FREQUENCY RANGE**
  - **EFFECTS OF LEVEL**
- **NOISINESS OF COMBINED SOURCES**

MULTIPLE EVENTS

- **DURATION/AMBIENT INTERACTION**
- **NUMBER OF OPERATIONS**
- **COMPARISON WITH JET AND PROP CTOL**

Figure 13

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N84 29676

HELICOPTER ENGINE CORE NOISE

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SUMMARY

Calculated engine core noise levels, based on NASA Lewis prediction procedures, for five representative helicopter engines are compared with measured total helicopter noise levels and ICAO helicopter noise certification requirements. Comparisons are made for level flyover and approach procedures. The measured noise levels are generally significantly greater than those predicted for the core noise levels, except for the Sikorsky S-61 and S-64 helicopters. However, the predicted engine core noise levels are generally at or within 3 dB of the ICAO noise rules. Consequently, helicopter engine core noise can be a significant contributor to the overall helicopter noise signature.

INTRODUCTION

In order for the United States to be in the best competitive world position, efficient low noise helicopters must be developed. Uncertainty in the prediction of noise and its control leads to poor performance/noise trades and overly conservative noise design margins that lead to economic penalties. Noisy helicopters can lead to night curfews at airports and heliports, as well as expensive suits by individuals and communities. Also the number of flights during daylight hours can be limited by the use of accumulative noise indices. Finally, helicopter noise can limit certain military stealth operations.

As part of a general program to alleviate community noise problems, the noise associated with helicopters has been receiving increased attention in recent years as the number of operating helicopters has multiplied. Studies, such as that in reference 1, have established that the most objectionable helicopter noise is related to the impulsive and non-impulsive noise generated by the main and/or tail rotors (fig. 1). Because of their dominance, these noise sources have relegated other helicopter noise sources to a secondary position. Consequently, such sources as engine noise and its potential to affect compliance with proposed and future civil helicopter certification requirements have been neglected. In addition to these major helicopter noise sources, noise is also generated by the interaction of the main rotor wake with the fuselage, external protuberances (pods, landing gear, etc.), and the tail rotor.

Effective measures to reduce helicopter noise require independent studies of each noise source in order to ascertain its contribution to system noise,and then a total system noise assessment. Once rotor noise has been reduced to acceptable levels, engine noise sources are the most significant noise generators (ref. 2). These sources consist broadly of compressor noise, core noise, and jet noise. Of these, core noise appears to be the most important engine noise source. The compressor generates high frequency source noise that can be effectively reduced by suitable blading design and acoustically treatment of the inlet duct surfaces. Currently, jet noise is not considered a major noise source for helicopters because of the low jet exhaust velocities. With low jet exhaust velocities, however, core noise can constitute the major engine noise source (ref. 3), and if sufficiently high can cause community annoyance. Core noise is difficult to suppress by wall acoustic treatment because it is dominated by a low frequency combustion noise component.

In the present paper, predicted core noise levels associated with helicopter operation will be examined to determine their significance in complying with the International Civil Aviation Organization (ICAO) helicopter noise certification requirements.

BACKGROUND

Core Noise

Core noise is considered to consist of that generated by the combustor, turbine, support struts, and internal surfaces. Combustor noise is produced by the unsteady combustion in turbine engines (ref. 2). That is, combustion is unsteady with time, which varies heat release and produces unsteady pressure fluctuations within the engine. These then propagate downstream from the combustor and give rise to a sound field. The sound field generated by the combustion process is partly attenuated by the turbine, depending on the number of stages, and to a lesser degree by the exhaust nozzle.

Reduction of the unsteady flow (turbulence) in a combustor in order to reduce the source noise may not be practical, since the combustion process depends on a high turbulence level for flame stability and burner performance optimization (ref. 2). Consequently, a performance penalty could be expected with reduced combustor noise.

Turbine noise sources are associated with high frequency generating mechanisms. Thus, tailpipe acoustic wall treatment, in principle, could suppress any objectionable turbine tones or noise levels. However, interactions between the turbine generated noise and the turbulent exhaust flow can result in increased overall noise levels (ref. 2).

Strut or obstruction noise is caused by the flow over a solid surface, resulting in a broadband noise source. In general, the flow velocities are sufficiently low within the engine boundaries that this noise source is considered a second order source. When strut noise does become apparent, it is generally caused by cross flow or rotating flow over an internal support member.

ICAO Helicopter Noise Certification Requirements

As part of the ICAO noise certification requirements for aircraft, a noise rule for helicopters has been adopted (ref. 4). The ICAO noise rule is substantially identical to the recently withdrawn FAA proposed rule (refs. 5 and 6). The following sections summarize the flight paths and noise measuring stations and the proposed noise rule.

Flight paths and noise measurement stations. - The proposed helicopter noise certification flights would consist of approach, level flyover, and takeoff noise tests. Simultaneous measurements for each noise test series would include a flight path noise measuring station and two sideline noise measuring stations, one on each side of the path and at a sideline distance of 150 m. The height of the helicopter over the noise measuring station is referenced to the flight path. A six degree angle (flight path) is proposed for the approach test, with a vehicle altitude of 120 m when the helicopter is directly over the flight path noise measuring station. For level flyover, a vertical height of 150 m is proposed for the vehicle flight path over the flight path noise measuring station. Finally, for the takeoff noise test, the measuring station is proposed to be located 503 m from the point at which takeoff power is applied in order to permit the measurement of the noise levels of the helicopter while at the best rate-of-climb attitude at high engine power and rotor settings.

For the level flyover tests, the reference speed proposed is 90 percent of either maximum level flight speed with maximum continuous power or the never-exceed speed, whichever is lower. The microphones would be located 1.2 m above the ground.

Noise rule. - The ICAO helicopter noise certification requirements (ref. 4) are given in figure 2. In general, the noise level varies with 10 log W for all certification requirements, the exception being at the low end of the gross weight scale. For approach noise, which has the highest allowable levels, the proposed noise limits vary between 87 and 107 EPNdB. For level flyover noise, which has the lowest allowable levels, the proposed noise limits vary between 85 and 105 EPNdB. The takeoff noise limits fall halfway between the two preceding sets of limits.

ACOUSTIC DATA BASE

In the present study, measured helicopter total noise data from reference 1 are used for comparison with predicted core noise levels and ICAO helicopter noise certificaton requirements. In reference 1, the measured helicopter noise levels are given for eight helicopters, two of which were powered by piston engines. These latter data are not included herein, the present study being limited to turbine engine powered helicopters. A brief description of the helicopter/engines included herein is given in the following table.

Helicopter	Engine	No. of engines	Test gross weight, W, kg
Hughes 500C	Allison 250-C20A	1	839
Bell 212 (UHIN Huey)	Pratt and Whitney PT6T-3	2	4354
Sikorsky S-61 (SH-3B)	General Electric T58-GE-8B	2	8492
Sikorsky S-64 (CH-54B)	Pratt and Whitney JFTD-12A-5A	2	19456
Boeing Vertol 114 (CH-47C, Chinook)	AVCO-Lycoming T55-L-11	2	18594

In addition to the preceding helicopters, a Bell 206L was also included in reference 1; however, the approach flight path was not the same as that for the other helicopters. Consequently, this set of data is not included herein. It should be noted that the Bell 206L is very similar to the Hughes 500C, having a somewhat larger gross weight (1768 kg) and a slightly more powerful Allison engine (20 more shaft horsepower).

The purpose of the tests of reference 1 was to obtain a data base for the development of the regulatory standards. Consequently, acoustic data were obtained at the measuring stations for approach and level flyover certification requirements. Data were also obtained at a hover condition (wheels 1.53 m above ground level), but are not included herein. No data were obtained for the takeoff condition. For the approach condition, glide slopes of 3°, 6°, and 9° were used for the acoustic measurements. Because the proposed certification requirements specify a 6° glide slope, only these acoustic data are included herein.

The following approach and level flyover helicopter airspeeds given in reference 1 were used herein:

Manufacturer	Nominal Airspeed, m/s	
	Approach	Level flyover
Hughes 500 C	26.8	58.3
Bell 212	30.8	56.7
Sikorsky S-61	30.8	59.3
Sikorsky S-64	30.8	49.0
Boeing Vertol 114	30.8	72.7

For both flight conditions, the helicopter engine centerline was assumed to be parallel to the ground.

ENGINE CHARACTERISTICS

The helicopter nominal full-power engine characteristics are given in the following table:

Engine	Maximum combustor-to-ambient pressure ratio, $P_{3,m}/P_{3,m}$	Maximum temperature ratio across combustor, $T_{4,m}/T_{3,m}$	Maximum combustor mass flow, $\dot{m}_m$, kg/sec
Allison 250/C20A	7.0	2.36	1.54
Pratt and Whitney PT6T-3	7.2	2.60	2.90
Gen. Electric T58-GE-8B	8.2	2.29	5.70
Pratt and Whitney JFTD-12A-5A	6.8	2.90	23.0
AVCO Lycoming T55-L-11	8.5	2.25	10.24

In the absence of data, input into noise prediction procedures for part-power engine operation for these engines was based on small turbofan engine operating characteristics (refs. 3 and 7) as described in reference 6.

CORE NOISE PREDICTION

Spectra

The spectral shape used for the prediction of core noise is that given in reference 8 and identified as the "spectral envelope". This spectral envelope is a broader spectrum than that frequently ascribed to combustor noise only. The peak of the spectrum is assumed to be at 400 Hz statically and is assumed to be shifted in flight by a Doppler shift in frequency.

Overall Sound Pressure Levels

The noise level statically is obtained from reference 9 and is given by:

$$\mathrm{OASPL}_{120^\circ} = K - 20 \log R + 10 \log \left\{ \dot{m}[(T_4 - T_3)(P_3/P_a)(T_a/T_3)]^2 \right\} \qquad (1)$$

where K, in SI units, is assumed to be 51 for turboshaft engines. The K value used for turboshaft engines is the average for turbojet and turbofan engines, as suggested in reference 8. The value of R is the azimuthal distance from the helicopter to the ground at each directivity angle. The variation of OASPL with directivity angle, taken from reference 9, is given in figure 3. The values shown are dB values relative to the OASPL at $\theta = 120^\circ$; this angle generally is considered to be the peak core noise angle.

A 3-signal coherence technique whereby the engine core noise can be directly measured is given in reference 10. The technique requires that fluctuating pressures be measured in the far-field and two locations within the engine core. The cross spectra of these measurements are used to establish the levels of the far-field noise that propagated from the engine core. The technique can be used even when other sources, such as jet noise, dominate. As reported in reference 10, the technique was applied to an AVCO Lycoming YF102 turbofan engine. A comparison of measured with predicted spectra is shown in figure 4 for engine speeds of 30 and 95 percent. Within the limitations of the measurements, reasonable overall agreement is noted between the measured and predicted spectra.

The YF102 engine core fits into the general category of engine core sizes frequently used with helicopters. On the basis of the preceding work, it is assumed herein that the predicted core noise spectral shape (ref. 8) and noise level (ref. 9) are applicable, as a first approximation, to helicopter engines.

In order to determine the flight effects from the static values of OASPL, the Doppler factor $(1 - M_0 \cos \theta)^{-1}$ was used in reference 9. The resultant inflight OASPL is given as follows:

$$\mathrm{OASPL}_F - \mathrm{OASPL}_S = -40 \log (1 - M_0 \cos \theta) \qquad (2)$$

Perceived Noise Levels

Perceived noise levels (PNL) were calculated for the appropriate engine power settings at approach and level flyover conditions. The PNL values, plotted as a function of time, were then integrated to a level 10 dB down from the peak PNL in order to obtain EPNL values for the various helicopters and flight conditions.

Calculated core noise levels were adjusted for the number of engines by adding 10 log N to the calculated single engine PNL and EPNL. An arbitrary 3 dB also was added to the calculated PNL and EPNL in order to account for ground reflections inherent in the measured data.

RESULTS

Perceived Noise Levels

In order to obtain the effective perceived noise levels (EPNL) for the core noise associated with the various helicopter engines, the predicted perceived noise levels (PNL) were plotted as a function of time before and after the overhead measurement station (figs. 5 and 6). This procedure is analogous to plotting the PNL as a function of distance along the flight path relative to the overhead measurement station.

Level flyover condition. - It is apparent from figures 5(a), (b), and (c) that the total noise levels, which are dominated by rotor noise (ref. 1), for the Boeing Vertol 114, Bell 212, and Hughes 500C greatly exceed the predicted core noise levels, even when the latter are for 100 percent core speed (maximum power). Upstream of the overhead measuring station (positive time), the predicted core noise levels at 100-percent core speed are near the measured total noise levels. This, however, may be coincidental, because the engine operating conditions for the tests are not available. Also shown in figure 5 are the predicted core noise levels for 91 percent core speed. It appears reasonable to assume that the measured noise levels were obtained in the range of 91- to 100-percent core speed.

For both the Sikorsky S-61 and S-64, the predicted core noise curve was very similar in shape to the trends in measured total noise levels; however, the measured levels were shifted toward more negative times compared with the predicted core noise curves. If the core speed had been near 86 percent of full core speed, the conclusions drawn between measured and predicted noise levels would be similar to those observed for the data shown in figures 5(a) to (c).

Approach conditions. - The measured total noise levels and predicted core noise level curves are shown in figure 6. In general, the trends of the variation of PNL with time for the approach condition are similar to those discussed for the level flyover condition.

Spectra

The spectra for the overhead measurement station are shown in figure 7 for both the level flyover and approach conditions. In general, the spectral data confirm the observations made in the discussion of the PNL trends. Because only one microphone was used for the measurements rather than an array along the flight path, the absolute measured spectral values are believed to be relatively less accurate than the calculated SPL curves. The measured data shown do not appear to be corrected for ground reflections, as evidenced by the large dips and rises in the spectra at the lower frequencies (<500 Hz). Consequently, the accuracy of the measured absolute SPL values are suspect.

COMPARISON OF CALCULATED CORE NOISE WITH ICAO HELICOPTER CERTIFICATION REQUIREMENTS AND MEASURED TOTAL NOISE

The ICAO helicopter noise rules for level flyover and approach conditions are shown in figure 8, together with the predicted core noise levels and the measured helicopter total noise levels.

Measured Total Noise Levels

The Hughes 500C, Bell 212, and Boeing Vertol 114 helicopters measured noise levels, shown by the circle symbols in figure 8, exceed the ICAO helicopter noise certification requirements, due to their high rotor noise components, by up to 5 EPNdB. The total measured noise levels for the Sikorsky S-61 and S-64 helicopters are below the proposed noise rule levels by as much as 4 EPNdB.

Predicted Core Noise Levels

Level flyover and approach. - The predicted core noise levels for the helicopters are also shown in figure 8 (square symbols). The core noise levels shown were calculated for a 100-percent core speed. In general, the level flyover predicted core noise levels are lower than the ICAO noise rule by about 3 EPNdB. The Sikorsky S-64 predicted core noise level, however, is 6 EPNdB below the proposed rule. For the approach condition, the predicted core noise levels are at the ICAO certification rule.

It should be noted that the predicted core noise levels and the measured total helicopter noise levels are substantially the same for the two Sikorsky helicopters. This can be due to several factors: (1) imprecise measured noise levels because of insufficient acoustic instrumentation (i.e., use of only a single microphone, etc.), (2) engines not operating at 100 percent core speed for the test flights; consequently, the predicted core noise levels should be lower than those indicated in figure 8, (3) the core noise correlation used from reference 9 may have to be modified to be applicable to helicopter turboshaft engines and (4) a combination of these factors.

It is apparent, however, that if the rotor noise is reduced, core noise constitutes a helicopter noise floor. Should future noise rules at even lower levels than those currently in effect be adopted - for example, consider proposed heliport levels near 85 EPNdB - core noise, in the absence of rotor noise, will provide a severe barrier toward achieving such a noise level, particularly for the heavier helicopters.

Takeoff condition. - The ICAO helicopter noise rule for takeoff prescribes a measuring station 503 m downstream of the initial start of climb. No altitude at the measuring station is specified because each helicopter has a climb rate depending, in part, on its weight and engine performance characteristics. Discussions with several representatives in the helicopter industry indicated that an altitude range of 100 to 200 m could be expected over the measuring station. This brackets the altitudes at the measuring stations for approach and level flyover (120 and 150 m, respectively). It was also indicated that, at the measuring station, a forward speed of about 41 m/s for helicopters is a good estimate. Consequently, the core noise levels for the takeoff conditions are on the same order as those for the approach and level flyover conditions. On these bases, it can be assumed that, in the absence of measured data, the relative differences between measured total noise levels and predicted core noise levels for takeoff are similar to those given for the approach and level flyover conditions.

FUTURE RESEARCH CONSIDERATIONS

In order to obtain more precise knowledge of helicopter engine core noise, a need exists to conduct core noise research with these engines. Several of the engines include reverse flow geometry in the hot section, as shown schematically in figure 9. Such a hot section configuration lends itself to thorough instrumentation with internal semi-infinite tube pressure probes (ref. 10). Probes could be located upstream of the combustor, within the combustor, between the combustor and turbine, between the high and low pressure turbine stages, and finally, downstream of the turbines. The hot section geometry also readily permits the study of the effect on combustion noise propagation of a variation in axial distance between the combustor and the turbine inlet by the insertion of various simple duct sections. Acoustic wall treatment as a means for combustion noise reduction could also be evaluated through the use of such auxiliary duct sections.

CONCLUDING REMARKS

On the basis of analytical calculation of core noise levels for current representative helicopter engines, it has been shown that, in general, core noise levels are within 3 EPNdB of the ICAO helicopter certification requirements. Because of an assumed constant applicable for turboshaft engines in the core noise prediction, the predicted core noise levels are valid only to ±5 dB. It is a strong possibility, however, that the core noise levels used herein are valid. Consequently, it is not improper to state that once rotor noise has been reduced to acceptable levels, engine core noise will provide a floor to further helicopter noise reductions.

The presence of a fuselage as a sound barrier between the engine noise sources and the ground has been advanced as an engine noise suppression device. However, studies of engine-over-the-wing concepts, such as reference 8, have shown that only high frequency noise is attenuated by the presence of a barrier. Thus, in the flyover plane, engine core and jet noise, which are low frequency noise sources, would not be attenuated significantly by the presence of a fuselage. In fact, the presences of a solid surface near a jet can result in low frequency noise generation or amplification. Compressor and turbine noise, being high frequency noise sources, would be reflected or shielded by a fuselage. The benefits of high frequency noise reduction by the shielding effects of a fuselage would not be evident at sideline locations.

In view of the preceding considerations, there is a need for research to be conducted to better understand the prediction and control of engine core noise which is a potentially important noise source relative to helicopter noise certification requirements.

SYMBOLS

c_a	ambient sonic velocity, m/sec
EPNL	effective perceived noise level, EPNdB
K	constant in internally-generated noise prediction, dB re 20 $\mu N/m^2$
$\dot{m}$	mass flow rate, kg/sec
M_0	flight Mach number, V_0/c_a, dimensionless
N	number of engines
OASPL	overall sound pressure level, dB re 20 $\mu N/m^2$
P	total pressure, N/m^2
PNL	perceived noise level, PNdB
R	source-to-observer distance, m
SPL	1/3-octave-band sound pressure level, dB re 20 $\mu N/m^2$
T	total temperature, K
V_0	flight speed, m/sec
W	gross takeoff weight
θ	directivity angle measured from inlet, deg

Subscripts:

a	ambient
F	flight
m	maximum
S	static
120°	evaluation at $\theta = 120°$
3	combustor inlet
4	combustor exit
θ	local directivity angle

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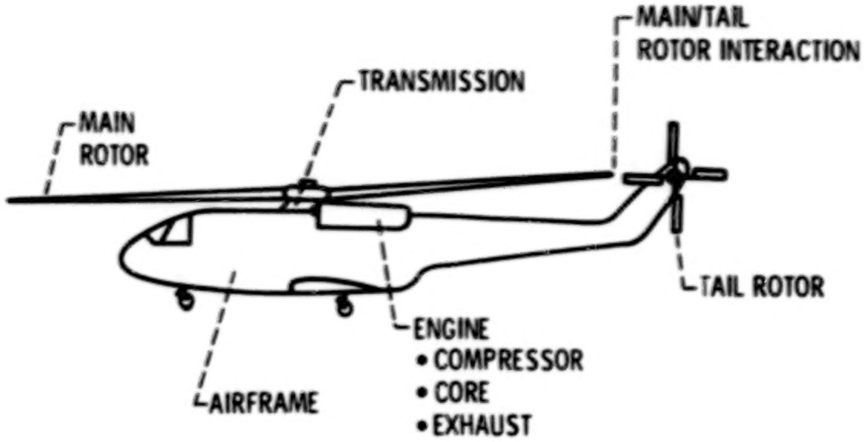

MAIN/TAIL
ROTOR INTERACTION
TRANSMISSION
MAIN
ROTOR
TAIL ROTOR
ENGINE
• COMPRESSOR
• CORE
• EXHAUST
AIRFRAME

Figure 1.- Helicopter noise sources.

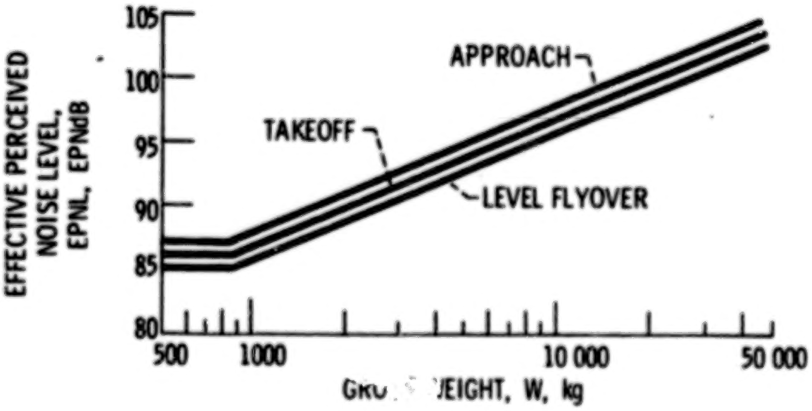


Figure 2.- ICAO helicopter noise certification requirements.

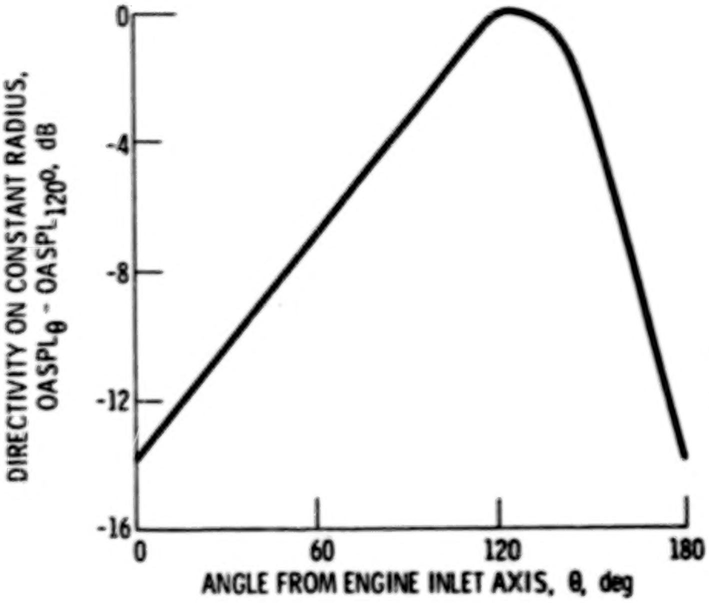
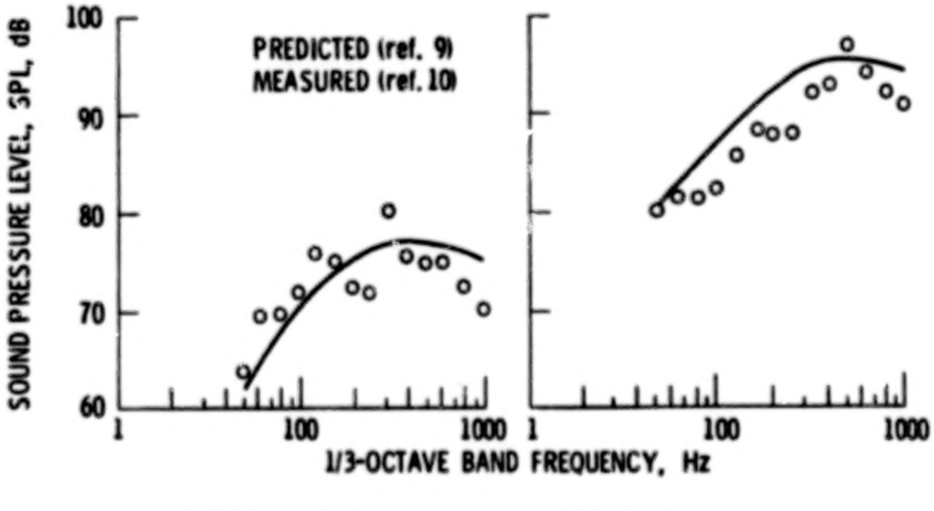
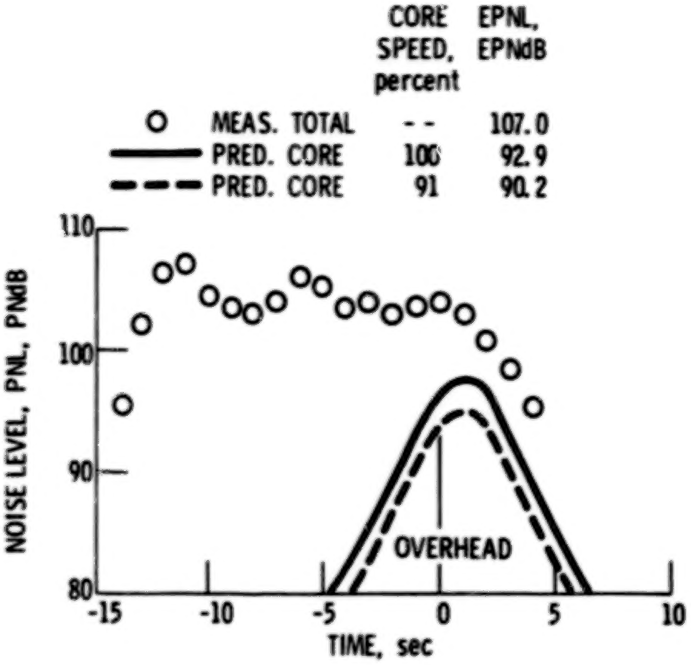


Figure 3.- Core noise static directivity (ref. 9).

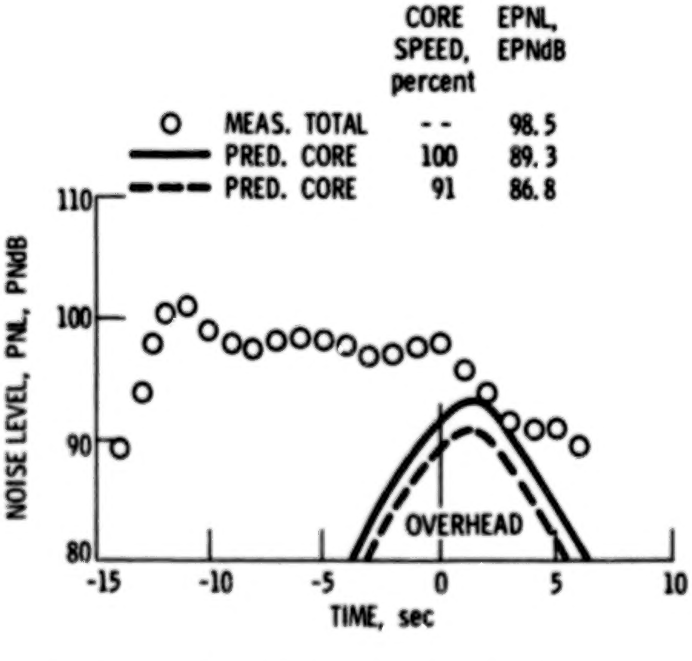


(a) 30% engine speed. (b) 95% engine speed.

Figure 4.- Comparison of measured and predicted YF-102 engine core noise spectra.

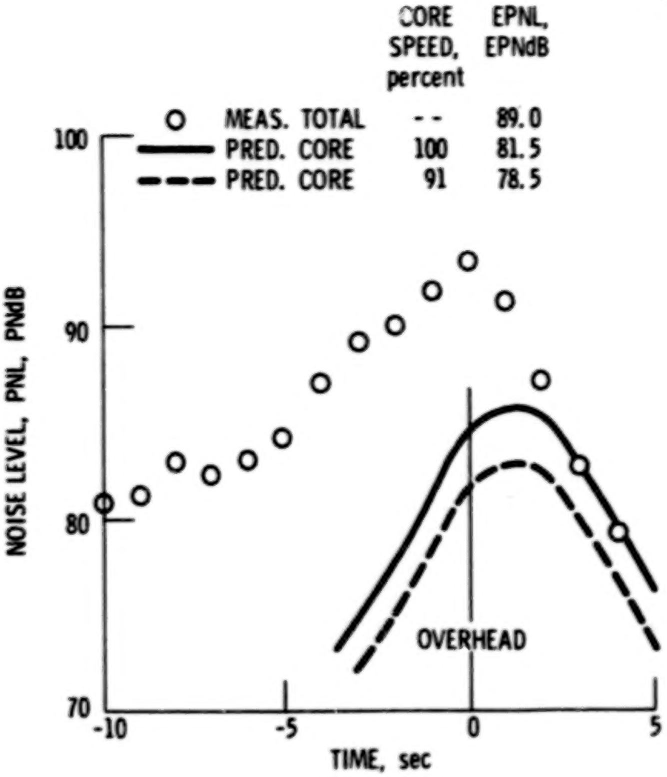


(a) Boeing Vertol 114; flight speed, 72.7 m/sec.

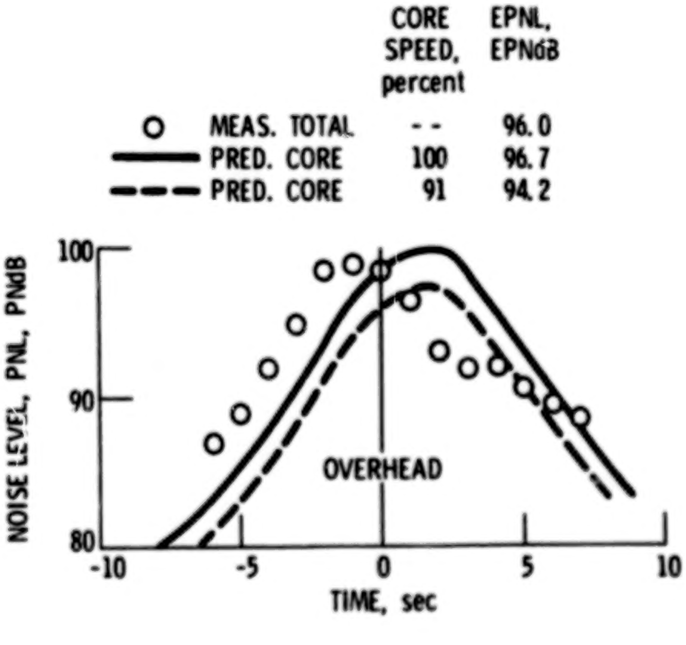


(b) Bell 212; flight speed, 56.7 m/sec.

Figure 5.- Variation of PNL with time for level flyover.

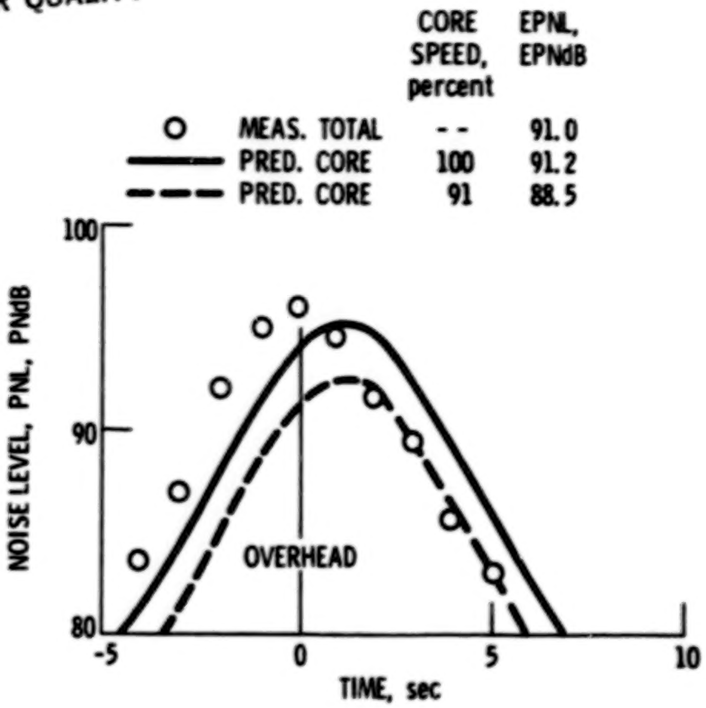


(c) Hughes 500 C; flight speed, 58.3 m/sec.



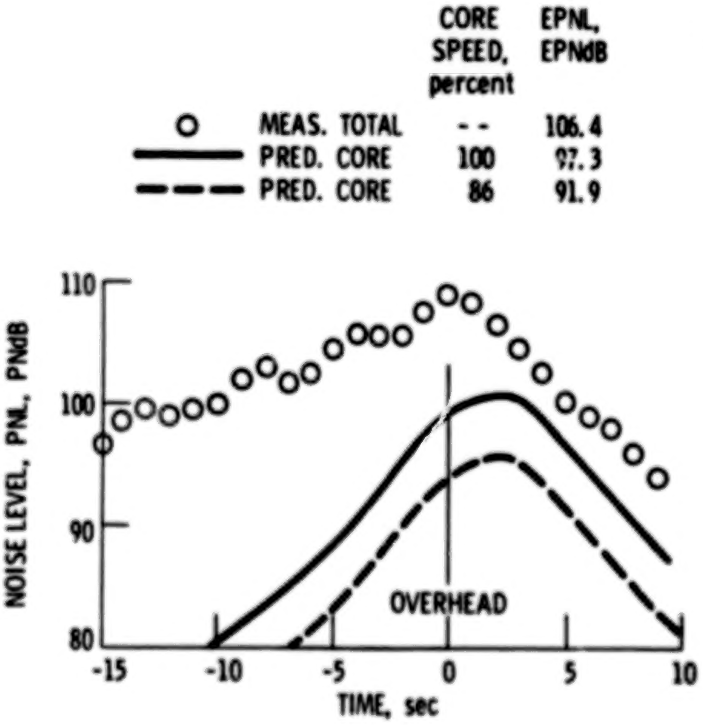
(d) Sikorsky S-64; flight speed, 49 m/sec.

Figure 5.- Continued.



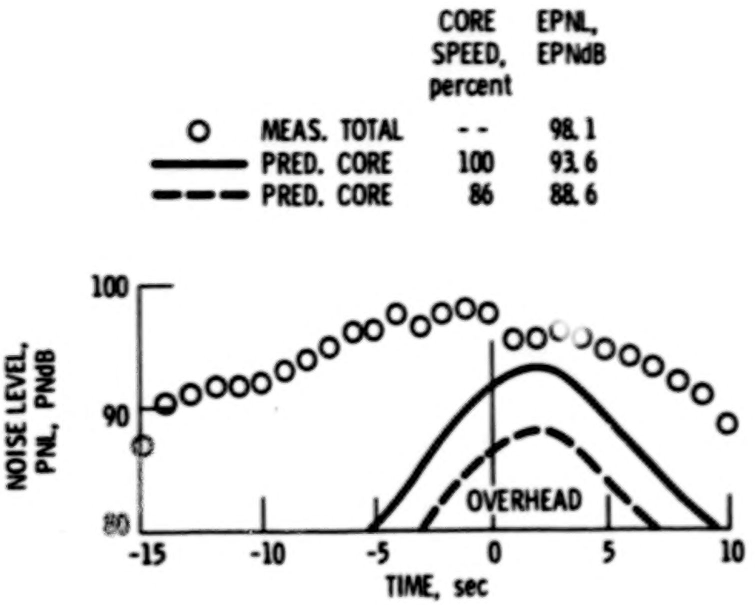
(e) Sikorsky S-61; flight speed, 59.3 m/sec.

Figure 5.- Concluded.

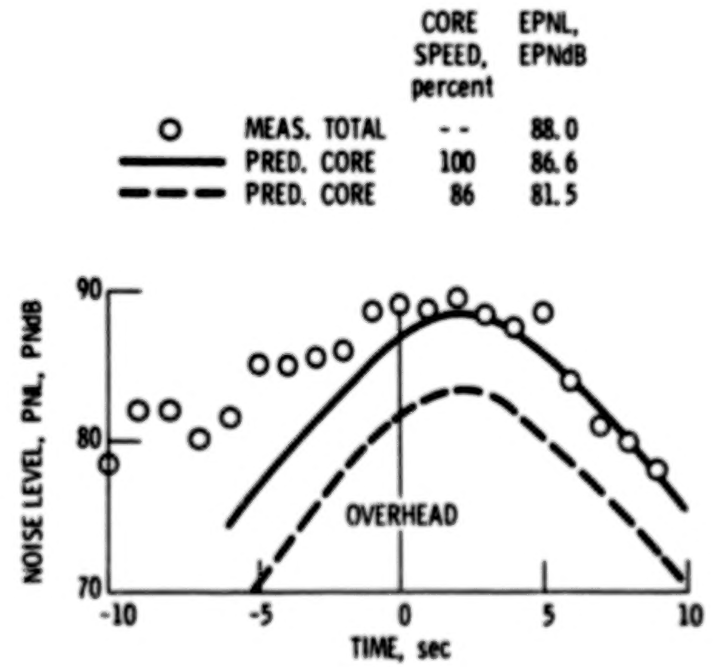


(a) Boeing Vertol 114; flight speed, 30.8 m/sec.

Figure 6.- Variation of PNL with time for approach condition.

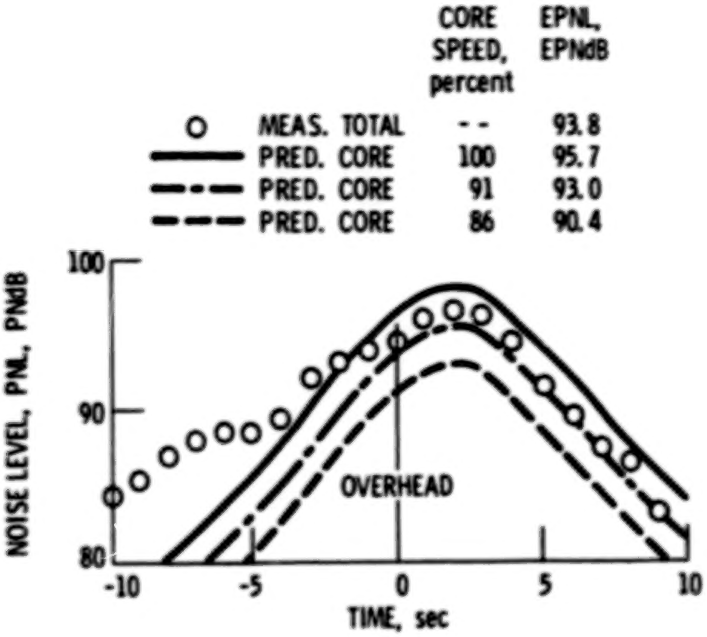


(b) Bell 212; flight speed, 30.8 m/sec.

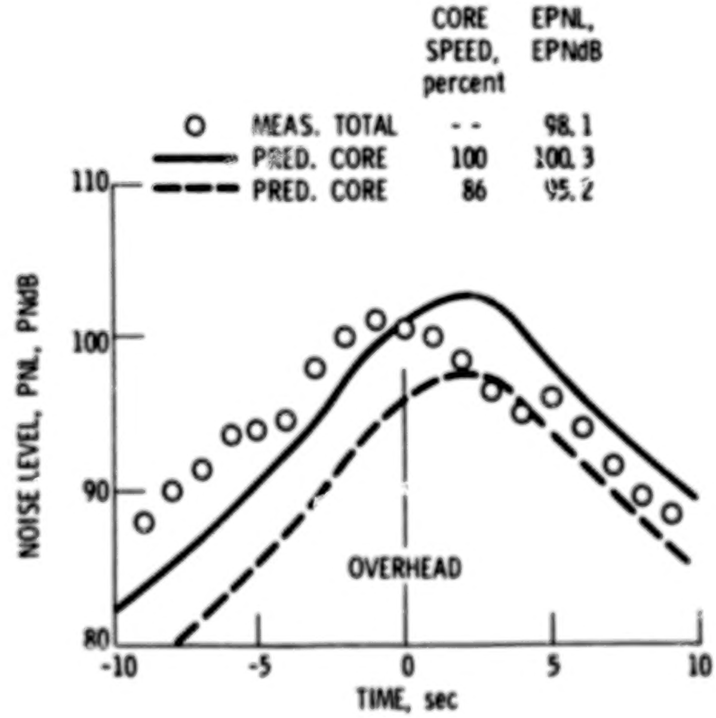


(c) Hughes 500 C; flight speed, 26.8 m/sec.

Figure 6.- Continued.

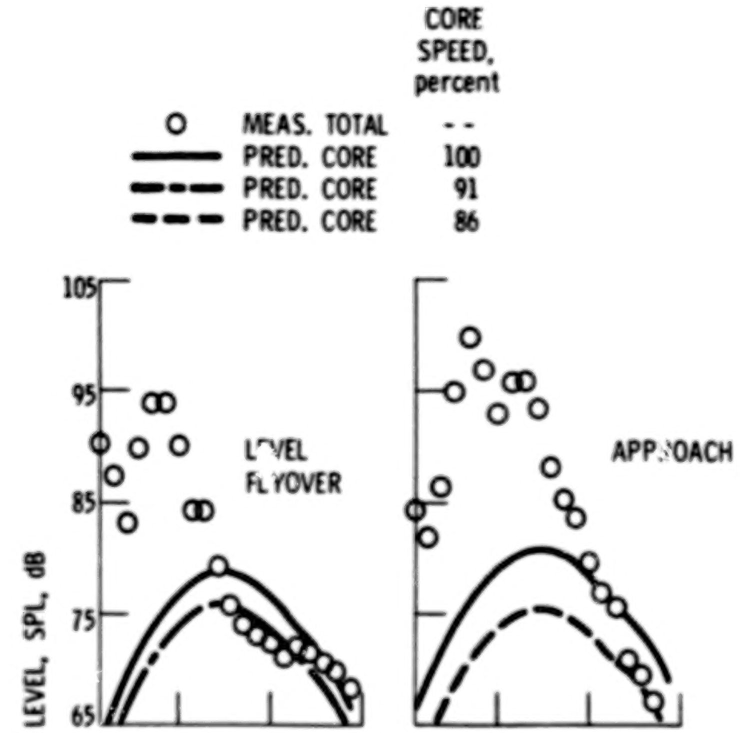


(d) Sikorsky S-61; flight speed, 30.8 m/sec.

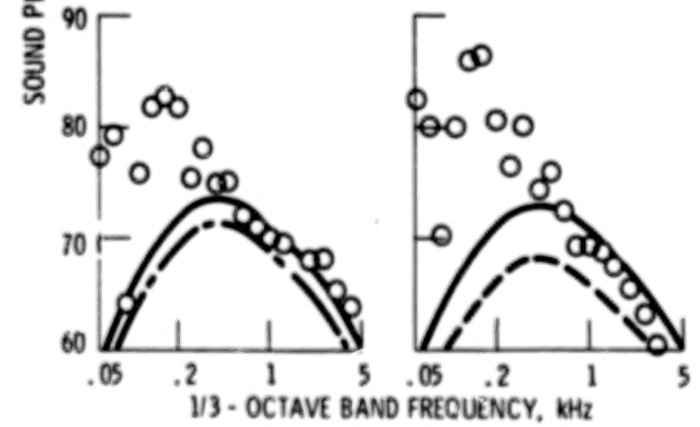


(e) Sikorsky S-64; flight speed, 30.8 m/sec.

Figure 6.- Concluded.

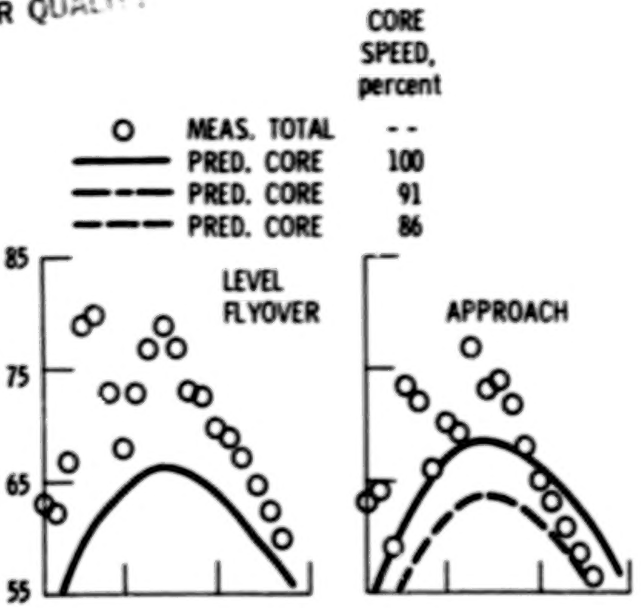


(a) Boeing Vertol 114.

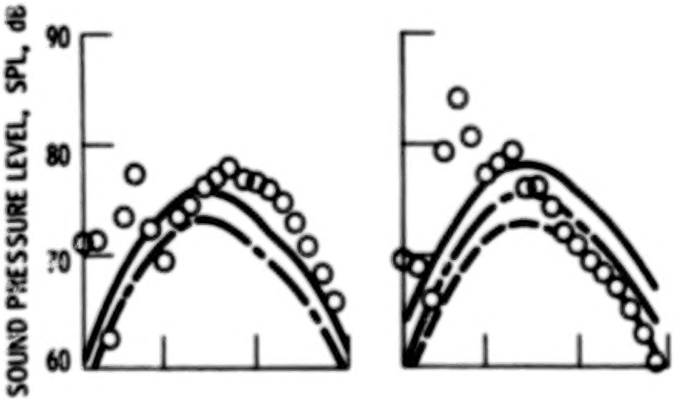


(b) Bell 212.

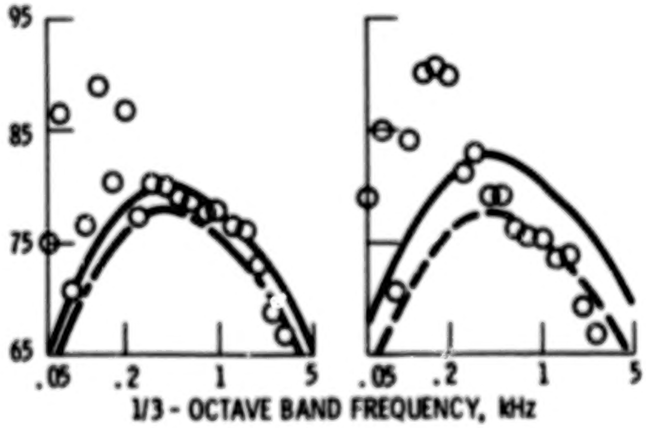
Figure 7.- Comparison of measured total helicopter noise spectra with predicted core noise spectra. Helicopter at overhead position.



(c) Hughes 500 C.

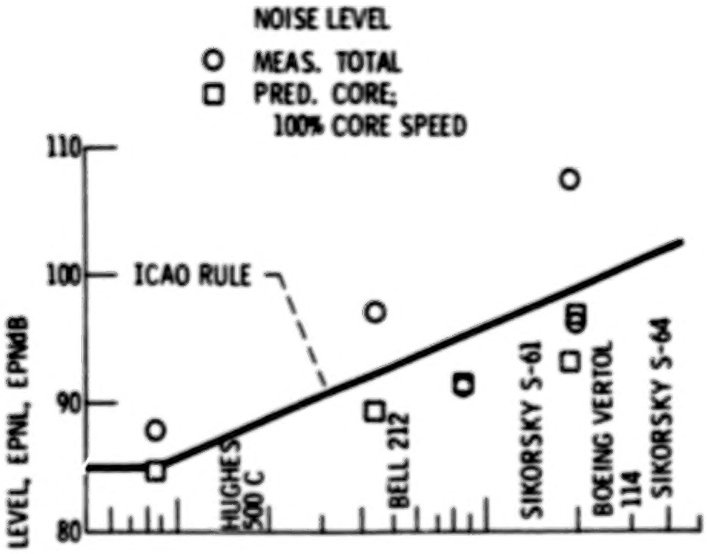


(d) Sikorsky S-61.

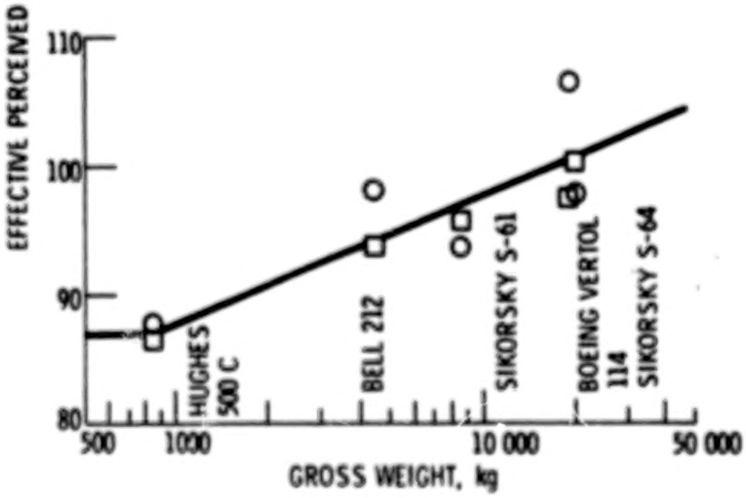


(e) Sikorsky S-64.

Figure 7.- Concluded.



(a) Level flyover.



(b) Approach.

Figure 8.- Comparison of ICAO helicopter noise certification requirements with measured total helicopter noise and predicted core noise levels.

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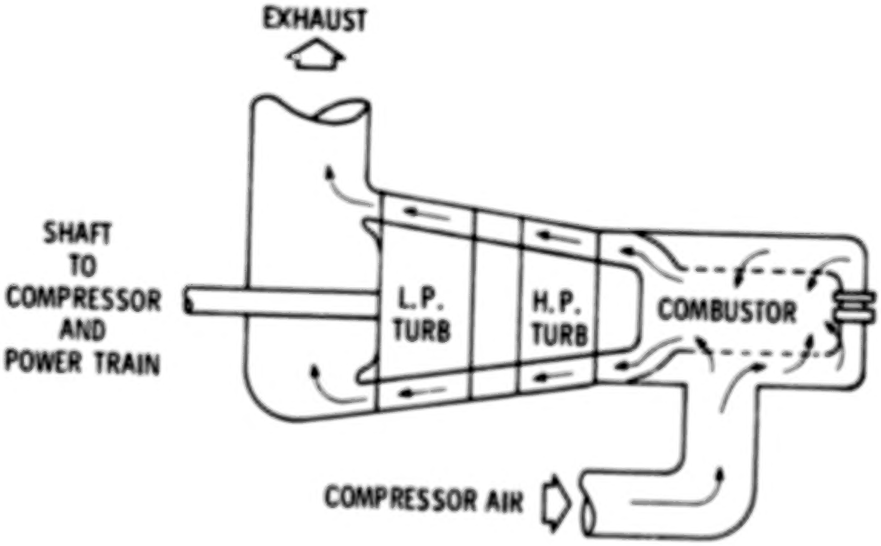
EXHAUST
SHAFT
TO
COMPRESSOR
AND
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H.P.
TURB
COMBUSTOR
COMPRESSOR AIR

Figure 9.- Schematic of helicopter engine hot section.

SESSION III

DEVELOPING AND VALIDATING NOISE PREDICTION METHODS

Chairman: J. P. Raney
NASA Langley Research Center

Vice-chairman: D. S. Jenney
Sikorsky Aircraft, United Technologies Corp.

INTRODUCTORY REMARKS

John P. Raney
NASA Langley Research Center
Hampton, Virginia

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INTRODUCTORY REMARKS

NASA/INDUSTRY GOAL: A VALIDATED DESIGN FOR NOISE PREDICTION CAPABILITY THAT ACCURATELY QUANTIFIES THE NOISE PRODUCED AT THE ICAO MEASURING POINTS BY AN EXISTING OR BY A PAPER AIRCRAFT.

We aren't there yet. The purpose of this session is to assess our progress and to determine our future directions.

To set the stage consider Figure 1, with which I have attempted to depict the complex nature of rotorcraft noise sources. Directivity and frequency spectrum are both extremely important ingredients in addition to acoustic power.

Figure 2 indicates a few of the contributing nonsteady loading sources of noise and their approximate frequency range.

Finally, note the complex nature of atmospheric and ground effects depicted in Figure 3 which can alter free-field noise levels by several decibels. A continuing research program at Langley has focused on quantifying these effects for CTOL aircraft[1,2]. Experiments with rotorcraft are planned.

Here at Langley we have built a prediction capability for jet-powered CTOL aircraft which incorporates much of the methodology also required for rotorcraft noise prediction[3,4,5]. For example, atmospheric propagation and ground effects, a variety of noise metrics, and a contouring algorithm are operational. Work on incorporating rotorcraft noise source modeling has commenced. Much has been done, and much remains to be done. A spirit of teamwork will speed our efforts and increase the probability of success.

The papers which follow evaluate progress on various aspects of rotorcraft noise prediction -- that is, the computation to an acceptable level of accuracy of EPNL at a specified location due to a flyby of a particular rotorcraft.

C - 4

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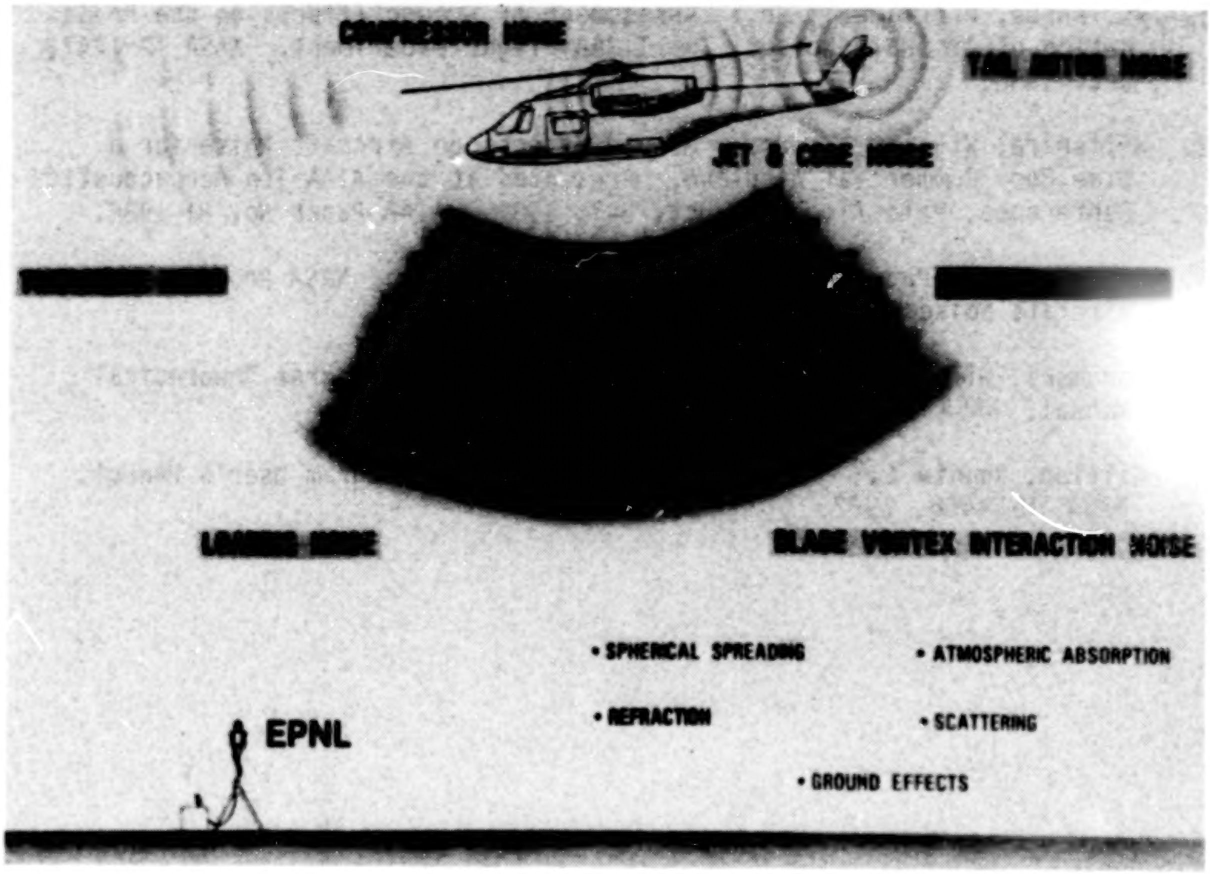
COMPRESSOR NOISE
TAIL ROTOR NOISE
JET & CORE NOISE
LOADING NOISE
BLADE VORTEX INTERACTION NOISE
• SPHERICAL SPREADING
• ATMOSPHERIC ABSORPTION
• REFRACTION
• SCATTERING
• GROUND EFFECTS
EPNL

Figure 1.- Helicopter noise modelling.

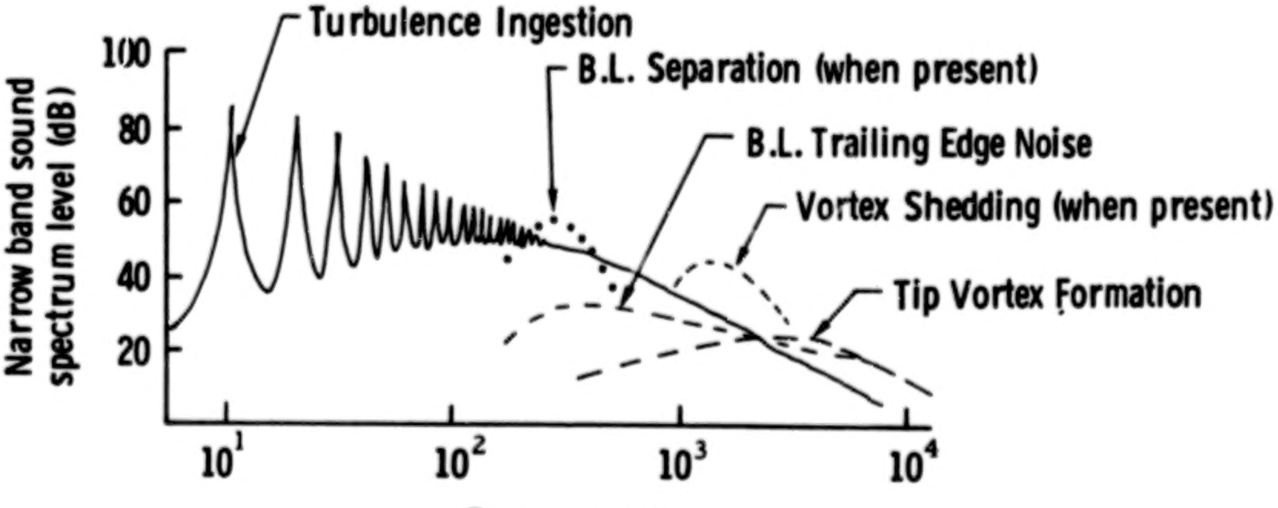

Figure 2.- Helicopter noise contributions from unsteady loading sources.

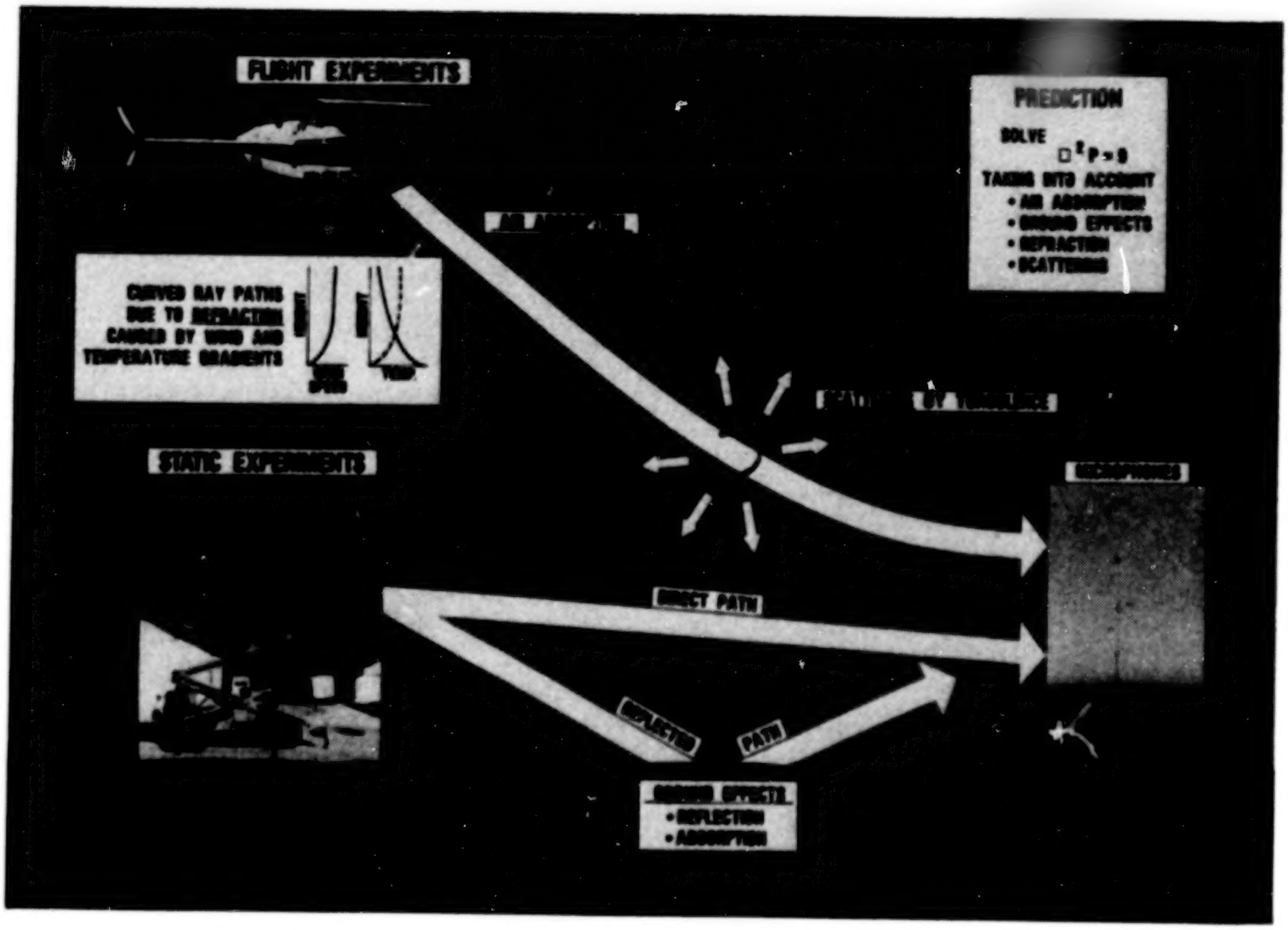
FLIGHT EXPERIMENTS
PREDICTION
SOLVE
$\Box^2 P = 0$
TAKING INTO ACCOUNT
• AIR ABSORPTION
• GROUND EFFECTS
• REFRACTION
• SCATTERING
AIR ABSORPTION
CURVED RAY PATHS
DUE TO REFRACTION
CAUSED BY WIND AND
TEMPERATURE GRADIENTS
HEIGHT
WIND SPEED
TEMP
SCATTERING BY TURBULENCE
STATIC EXPERIMENTS
MICROPHONES
DIRECT PATH
REFLECTED
PATH
GROUND EFFECTS
• REFLECTION
• ABSORPTION

Figure 3.- Acoustic propagation research.

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ROTORCRAFT NOISE PREDICTION

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Hampton, Virginia

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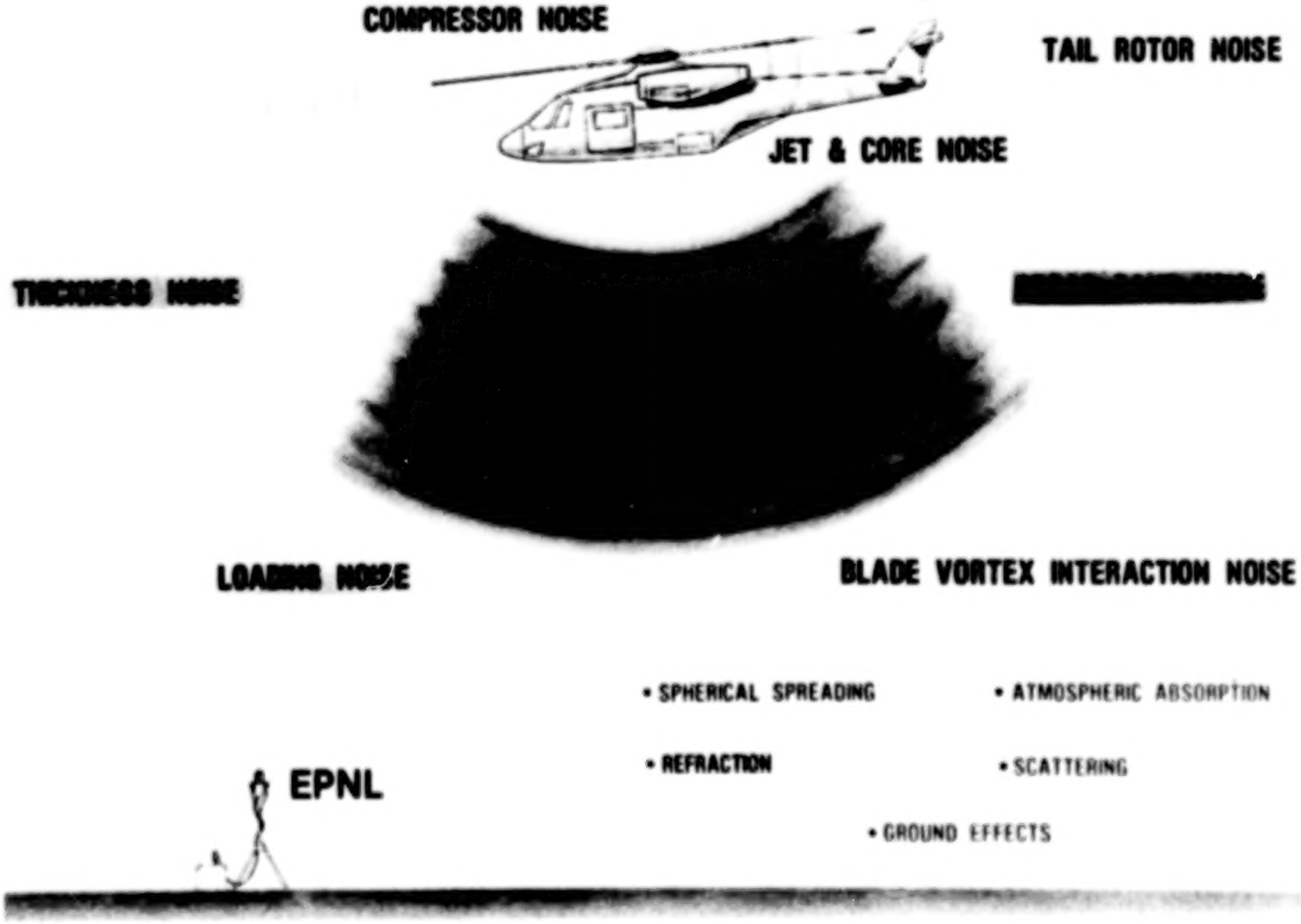
SUMMARY

The purpose of this presentation is to give a general strategy for rotorcraft noise prediction. This strategy is expressed through a modular software system design rather than theoretical analysis of the aeroacoustic phenomena. The crucial design choices in a software system design are the module interface definitions. An interface is the data that are passed from one module to another. A module takes data from one (input) interface and transforms it, through a prediction method, to another (output) interface. In system design, the method is less important than the interface. The two types of methods available may be broadly classified as empirical or analytical, although no method is purely one or the other. These two general approaches will be compared as they apply to rotorcraft noise prediction.

- STRATEGY FOR ROTORCRAFT NOISE PREDICTION
- MODULE INTERFACE DEFINITIONS
- COMPARISON OF EMPIRICAL AND ANALYTICAL METHODS

HELICOPTER NOISE MODELING

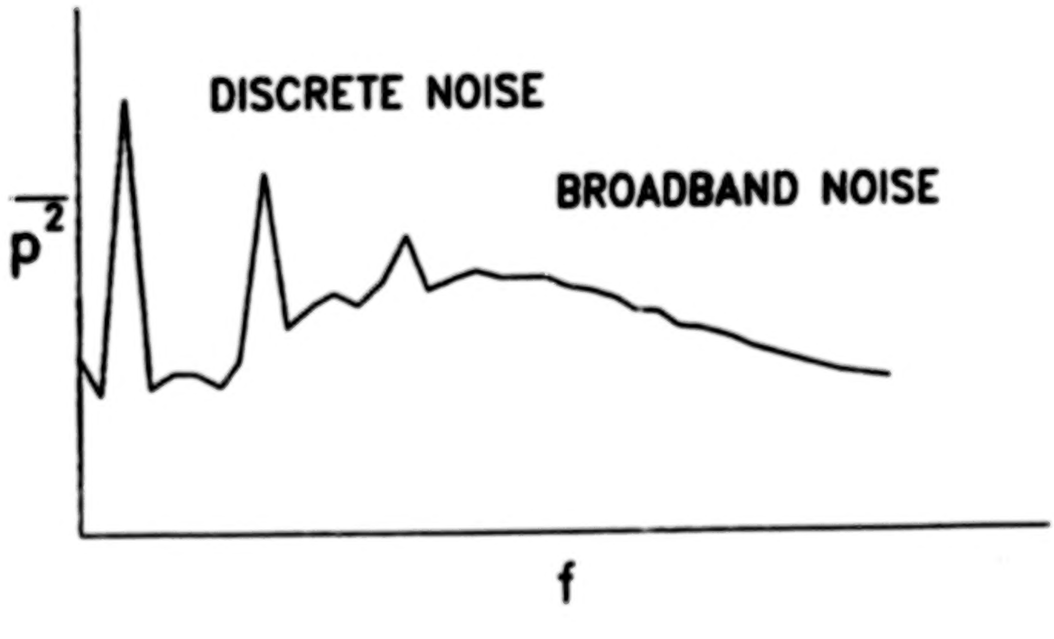
Helicopter noise modeling involves the description of acoustic source components such as the main rotor, tail rotor, and engine. These components may be further subdivided into subcomponents such as individual blades and into source types such as thickness and loading noise. The modeling also involves the propagation effects of the atmosphere and the ground and the integration of the received noise spectrum into measures of subjective response such as EPNL. A top-down software design process begins with the desired result, namely the received noise, and works back toward the source through an interlocked sequence of system interfaces and modules.


COMPRESSOR NOISE
TAIL ROTOR NOISE
JET & CORE NOISE
THICKNESS NOISE
LOADING NOISE
BLADE VORTEX INTERACTION NOISE
• SPHERICAL SPREADING
• ATMOSPHERIC ABSORPTION
• REFRACTION
• SCATTERING
• GROUND EFFECTS
EPNL

NOISE SPECTRUM REPRESENTATION

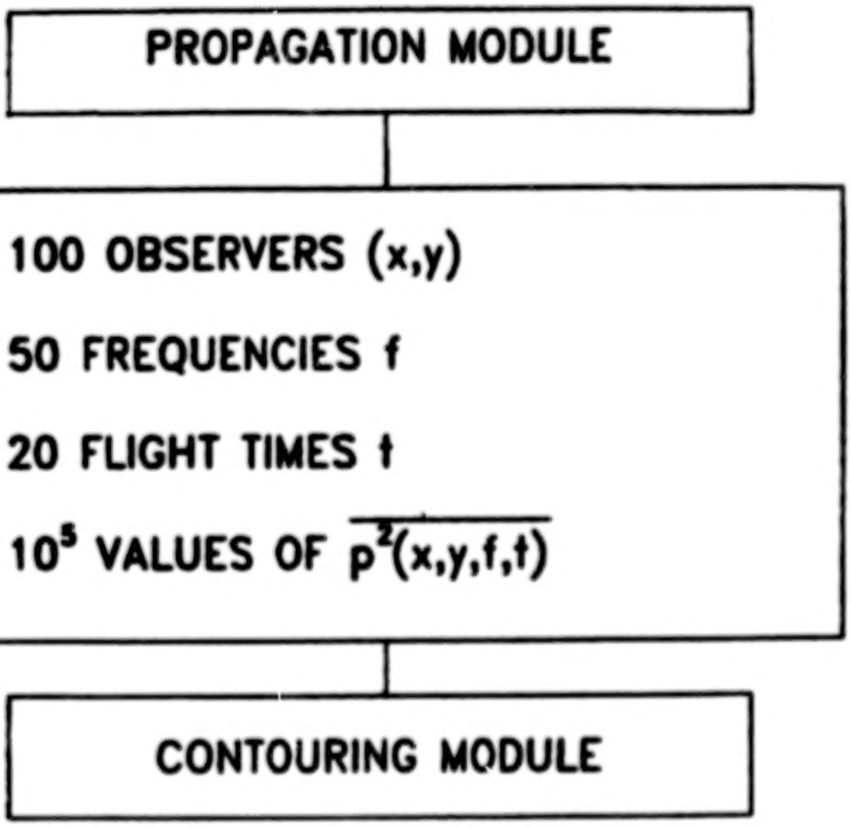
The first design choice is the representation of the noise. Helicopter noise is typically described in terms of a spectrum with a broadband component and superimposed tones, or discrete components. Alternative descriptions are narrowband spectra and time series of acoustic pressure.

$$\overline{p^2}(f) = \overline{p^2}_{bb}(f) + \Sigma\, \overline{p^2}_i\, \delta(f-f_i)$$



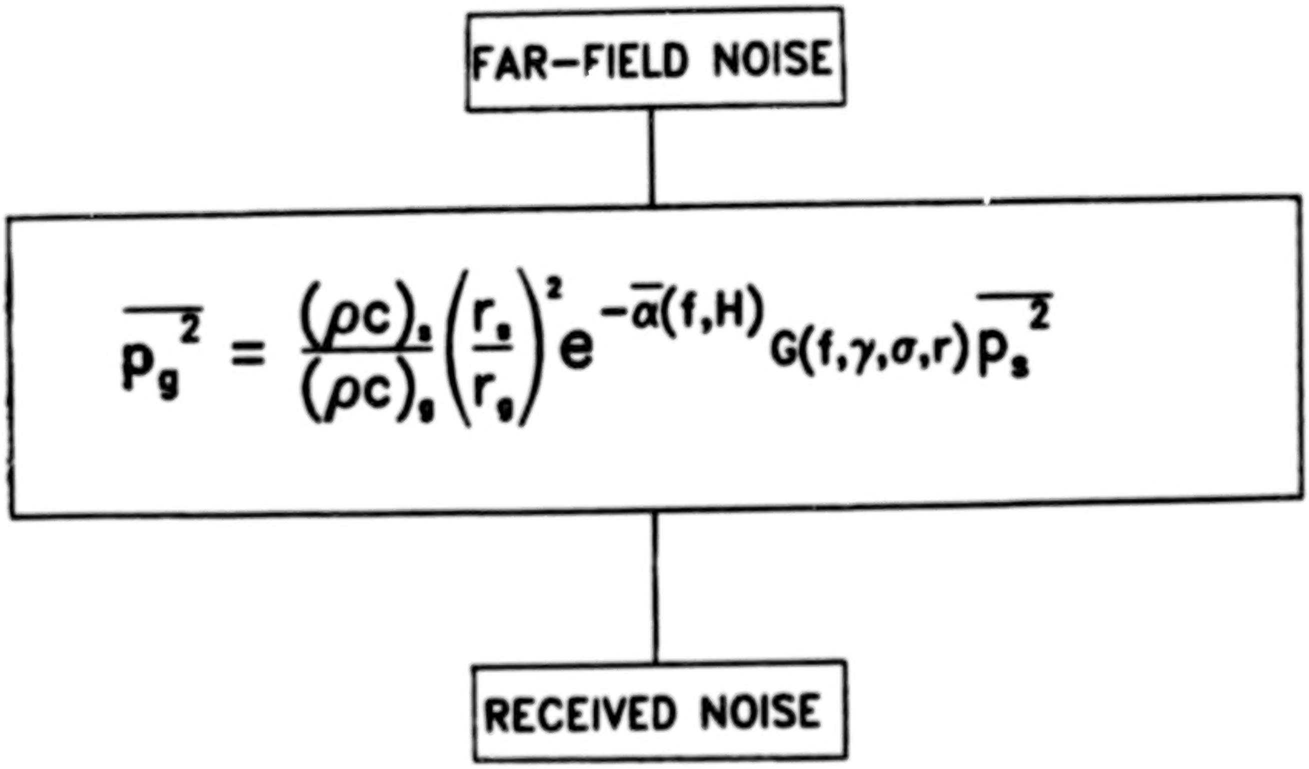
RECEIVED NOISE INTERFACE

The first interface (often considered last) is the received noise. A description of the noise from a helicopter operation may involve the time-dependent spectrum for about 100 observer positions in the community. These data, consisting of about 100,000 values of mean-squared pressure, may be integrated to form any subjective unit and used to generate noise contours.


PROPAGATION MODULE
100 OBSERVERS (x,y)
50 FREQUENCIES f
20 FLIGHT TIMES t
10^5 VALUES OF $\overline{p^2(x,y,f,t)}$
CONTOURING MODULE

PROPAGATION MODULE

The received noise data are supplied by a propagation module which transforms mean-squared pressures from a large sphere around the source (helicopter) to mean-squared pressures on the ground. Four effects are usually applied in this transformation. These are the change in atmospheric impedance, the spherical spreading, the atmospheric attenuation, and the ground effect. The ground effect is a very complex term which depends on frequency, incidence angle, ground resistance, and range.



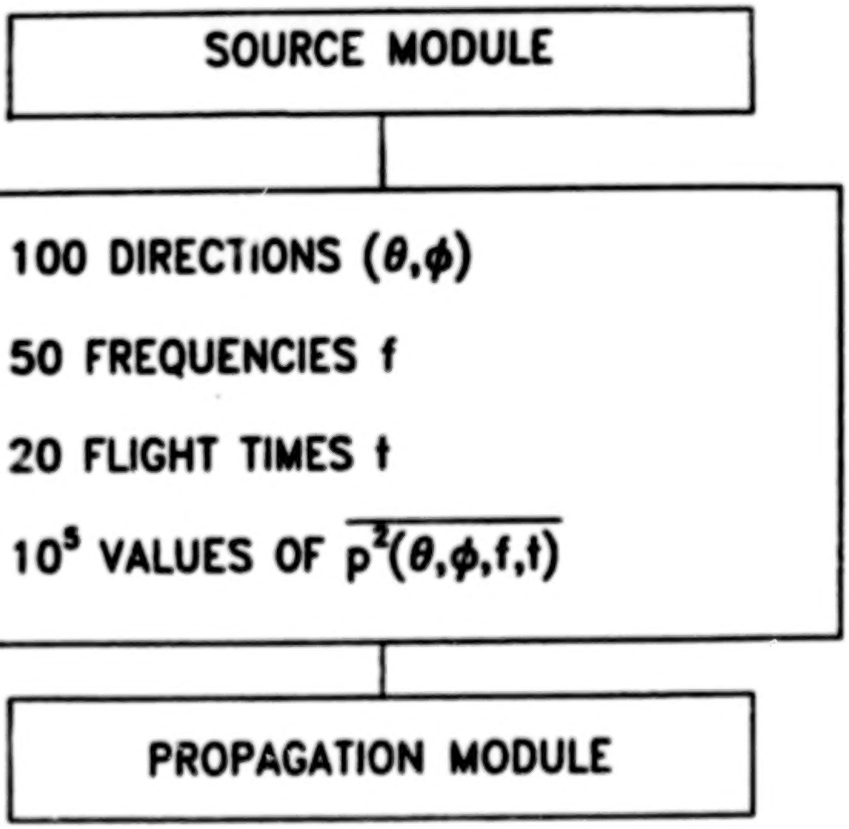
FAR-FIELD NOISE

$$\overline{p_g^{\,2}} = \frac{(\rho c)_s}{(\rho c)_g}\left(\frac{r_s}{r_g}\right)^2 e^{-\bar{\alpha}(f,H)}\, G(f,\gamma,\sigma,r)\,\overline{p_s^{\,2}}$$

RECEIVED NOISE

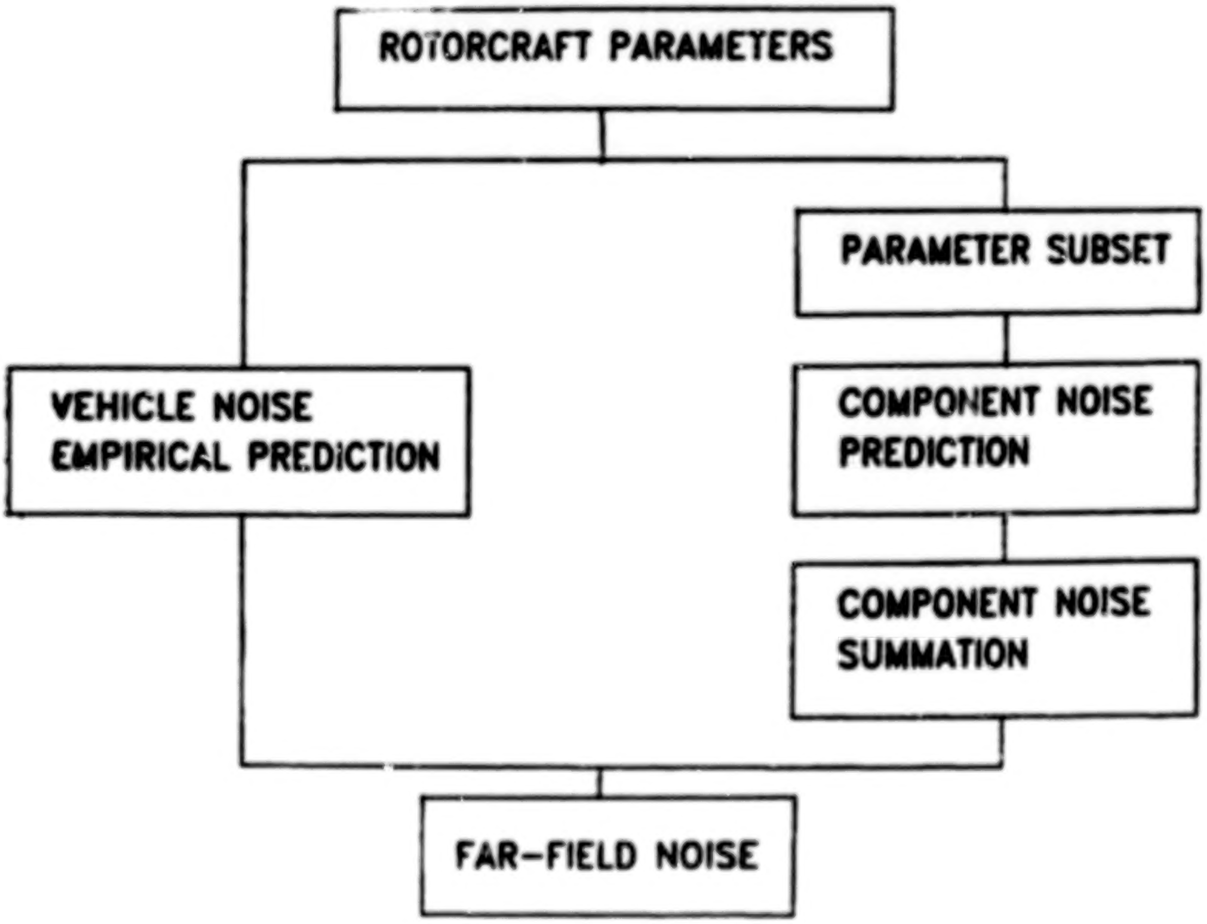
FAR-FIELD NOISE INTERFACE

The far-field noise interface is the mean-squared pressure on a large sphere several rotor diameters from the source. Time-dependent spectra are tabulated at about 100 different directions for input to the propagation module. This interface also consists of about 100,000 data elements.


SOURCE MODULE
100 DIRECTIONS (θ,ϕ)
50 FREQUENCIES f
20 FLIGHT TIMES t
10^5 VALUES OF $\overline{p^2(\theta,\phi,f,t)}$
PROPAGATION MODULE

ROTORCRAFT NOISE PREDICTION METHODS

A rotorcraft noise prediction method transforms the parameter interface into the far-field noise interface. Here, there is a fundamental choice. An empirical formula could be used to predict the far-field noise directly from the rotorcraft parameters. Subsets of the parameters can be used to predict noise for separate components such as the main and tail rotors. These components are then summed to give the total far-field noise.


ROTORCRAFT PARAMETERS
PARAMETER SUBSET
VEHICLE NOISE EMPIRICAL PREDICTION
COMPONENT NOISE PREDICTION
COMPONENT NOISE SUMMATION
FAR-FIELD NOISE

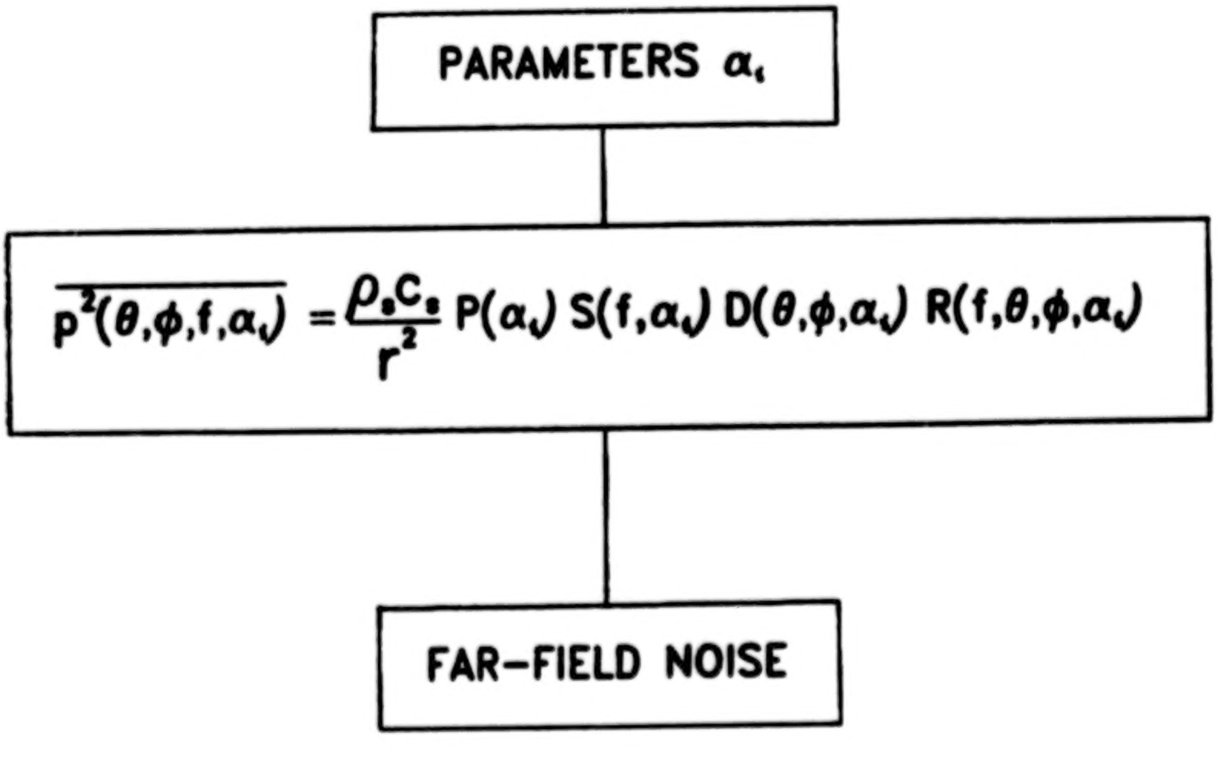
ROTORCRAFT PARAMETER INTERFACE

The rotorcraft parameter interface is the data which totally describe the state of the vehicle as a function of time. These include the atmospheric conditions, the flight path, the engine state variables, and the rotor characteristics.

- ATMOSPHERIC PARAMETERS
  - DENSITY
  - SOUND SPEED
- FLIGHT PARAMETERS
  - POSITION
  - VELOCITY
  - ORIENTATION
  - ANGULAR VELOCITY
- ENGINE STATE PARAMETERS
  - TOTAL PRESSURE
  - TOTAL TEMPERATURE
  - MASS FLOW RATE
  - FUEL/AIR RATIO
- ROTOR PARAMETERS
  - NUMBER OF BLADES
  - ROTATIONAL SPEED
  - PITCH SCHEDULE
  - HINGE CHARACTERISTICS
  - BLADE SHAPE, MASS, AND STIFFNESS

EMPIRICAL NOISE PREDICTION

Empirical noise prediction is based on measuring noise from a component and collapsing the measured data through curve fits in terms of the independent parameters. Any data set where mean-squared pressure is measured can be separated into four terms with increasing complexity (number of dimensions). The acoustic power depends only on the parameters. The power spectrum depends on frequency and the parameters. The overall directivity depends on two directional variables and the parameters. The relative spectrum, which relates pressure spectrum to power spectrum, is a function of all of the variables. Because of its complexity, many empirical formulas assume that the relative spectrum is unity; that is, spectral and directivity effects separate. The large number of parameters required to represent a rotorcraft makes empirical prediction of the entire vehicle inaccurate.



$$\overline{p^2(\theta,\phi,f,\alpha_i)} = \frac{\rho_0 c_0}{r^2} P(\alpha_i)\, S(f,\alpha_i)\, D(\theta,\phi,\alpha_i)\, R(f,\theta,\phi,\alpha_i)$$

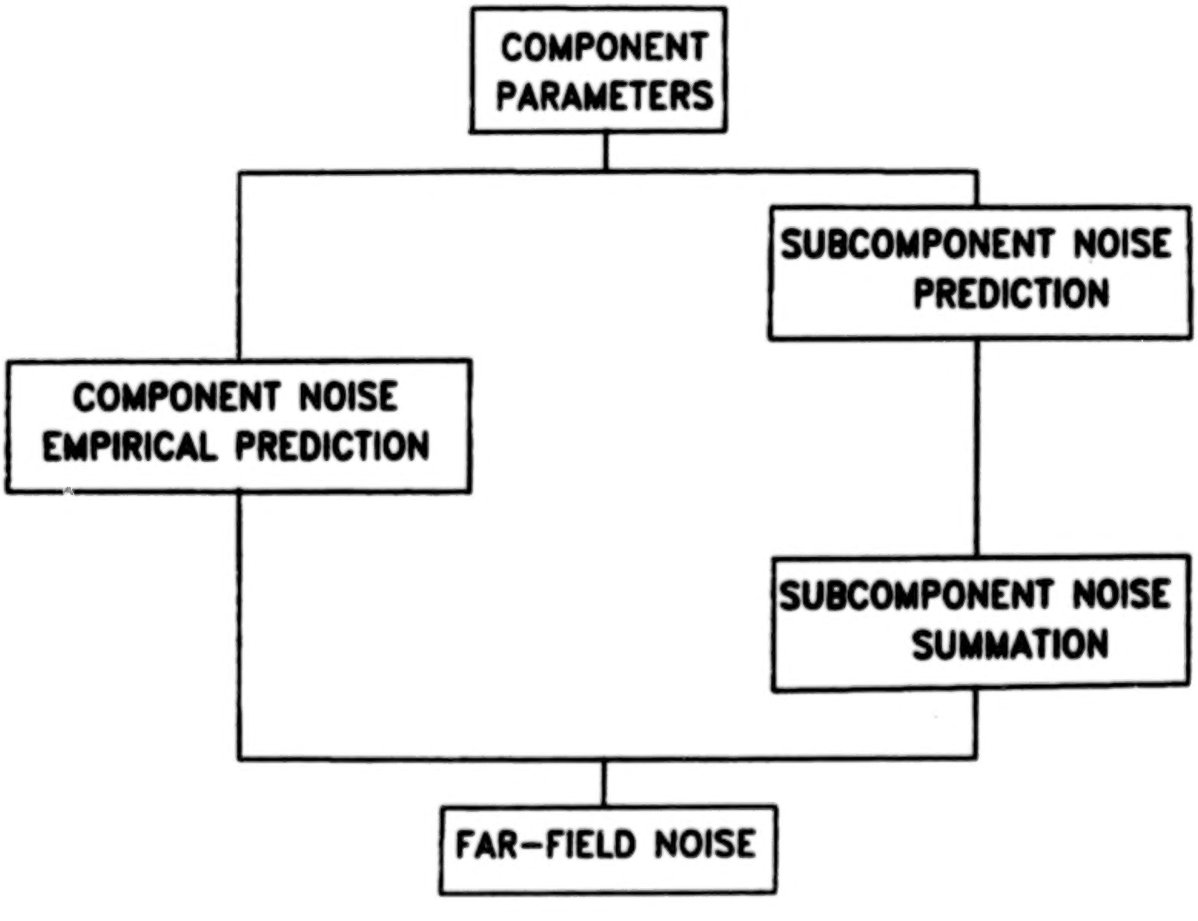
COMPONENT NOISE SUMMATION

Component noise summation is usually made using the mean-squared pressure assumption. Cross-covariance terms such as main and tail rotor, main rotor and engine, and engine and tail rotor are neglected. This assumption is justified by the fact that signals from these different sources will not maintain fixed phase relationships so that ensemble averaging, indicated by the overbar, leaves only the autocovariance terms in the summation.

$$
\begin{aligned}
\overline{p^2} = \; & \overline{p_M p_M} + \cancel{\overline{p_M p_T}} + \cancel{\overline{p_M p_E}} \\
& + \cancel{\overline{p_T p_M}} + \overline{p_T p_T} + \cancel{\overline{p_T p_E}} \\
& + \cancel{\overline{p_E p_M}} + \cancel{\overline{p_E p_T}} + \overline{p_E p_E}
\end{aligned}
$$

COMPONENT NOISE PREDICTION

Component noise prediction may be made using the previously described empirical formula or by predicting the noise of subcomponents and summing these. Empirical prediction is far more realistic for components than for the entire vehicle since the number of parameters is fewer and there is often an axis of symmetry. Empirical prediction of engine noise is based entirely on the axisymmetric assumption. The empirical method given here has been successfully applied to coaxial jet noise where there are five independent parameters, but this number of parameters is near the limit of where the empirical method will work. Subcomponent noise prediction may be empirical or analytical, but the summation process is unclear at this time.


COMPONENT PARAMETERS
SUBCOMPONENT NOISE PREDICTION
COMPONENT NOISE EMPIRICAL PREDICTION
SUBCOMPONENT NOISE SUMMATION
FAR-FIELD NOISE

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SUBCOMPONENT NOISE SUMMATION

The subcomponent noise summation process also involves a decision regarding cross-covariance terms. Considering a three-bladed main rotor, for example, analytic prediction usually assumes perfect phase relationships between blades whereas empirical prediction would probably assume a mean-squared summation rule. Since analytic methods tend to work at low frequencies, the cross-covariance terms may not be neglected; however, analytic methods fail at high frequencies which supports the empirical assumption. An accurate determination of cross-covariances of signals from different subcomponents is essential for improving the accuracy of rotorcraft noise predictions.

$$\begin{aligned}\overline{p_M p_M} = {} & \overline{p_{M_1} p_{M_1}} + \overline{p_{M_1} p_{M_2}} + \overline{p_{M_1} p_{M_3}} \\ & + \overline{p_{M_2} p_{M_1}} + \overline{p_{M_2} p_{M_2}} + \overline{p_{M_2} p_{M_3}} \\ & + \overline{p_{M_3} p_{M_1}} + \overline{p_{M_3} p_{M_2}} + \overline{p_{M_3} p_{M_3}}\end{aligned}$$

SUBCOMPONENT NOISE PREDICTION

Ffowcs Williams-Hawkings Equation

Subcomponent noise prediction may be done empirically but since this process has been described for components, analytical predictions will be assumed. One analytical method is to solve the Ffowcs Williams-Hawkings (FW-H) equation outside of some surface moving through the air. The free-space Green's function is used to transform this equation into Farassat's integral equation. When the quadrupole sources are neglected, this equation gives the acoustic pressure at any point in terms of the pressure and velocity on the given surface. The solution of this equation is dealt with in other papers and will not be considered here; however, it should be noted that the variables in Farassat's equation should be treated as random variables in order to properly account for the summation of subcomponent signals into component noise.

$$\overline{\Box}^2 p' = \mathfrak{M} + \mathfrak{D} + \mathfrak{Q} \; : \; f(x_i,t) \geq 0$$

$$\mathfrak{M} = \frac{\bar{\partial}}{\partial t}\left\{[\rho_\infty v_n + \rho(u_n - v_n)]|\nabla f|\delta(f)\right\}$$

$$\mathfrak{D} = -\frac{\bar{\partial}}{\partial x_i}\left\{[n_i p + \rho(u_n - v_n)]|\nabla f|\delta(f)\right\}$$

$$\mathfrak{Q} = \frac{\bar{\partial}^2}{\partial x_i \partial x_j}\left\{[\rho u_i u_j + \delta_{ij}(p - \rho' c_\infty^2)]H(f)\right\}$$

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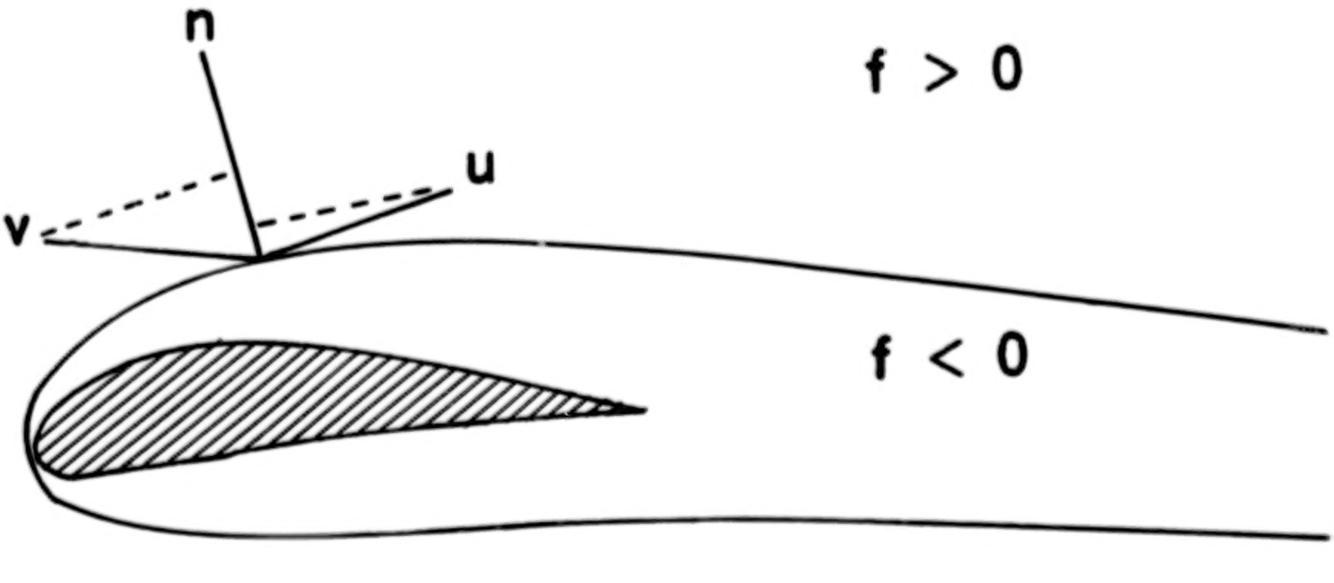
SUBCOMPONENT NOISE PREDICTION

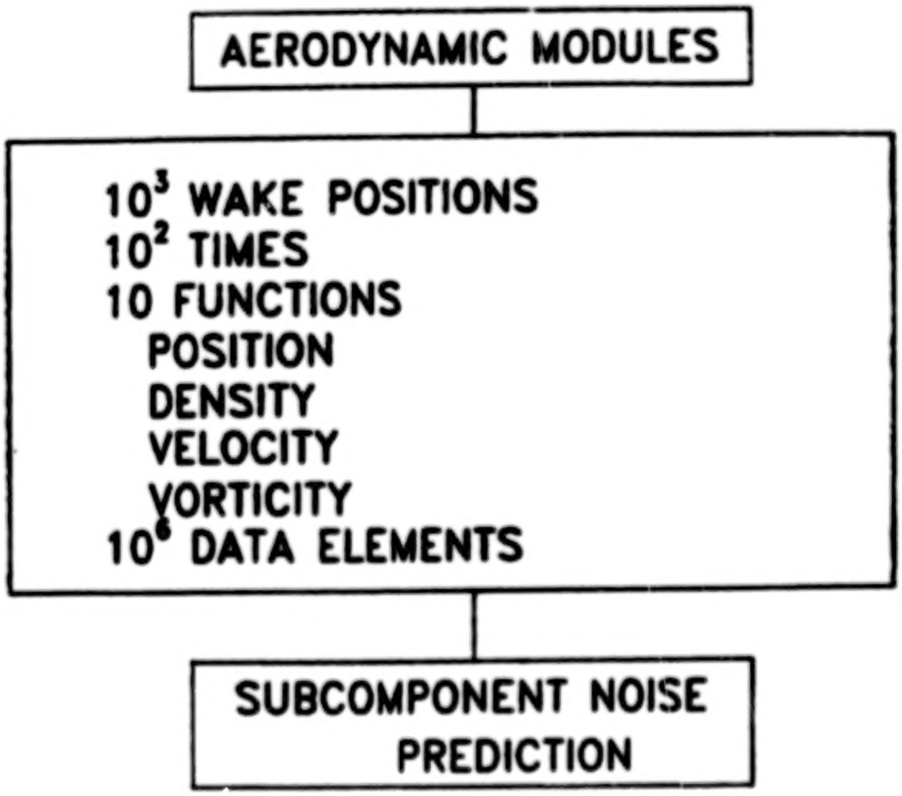
Boundary Layer/Wake Surface

The surface on which the integral is taken is not necessarily the blade surface. Quadrupole sources are not necessarily negligible in the boundary layer and the wake so that it may be necessary to move the surface out from the blade to the edge of the boundary layer and wake in order to transform the FW-H equation into Farassat's two-dimensional integral equation. In this case, the surface velocity and fluid velocity are unequal and there are additional monopole and dipole sources of noise. Farassat's integral equation becomes a boundary condition on the aerodynamic equations. The surface where Farassat's equation is applied will be called the aeroacoustic interface.



AEROACOUSTIC INTERFACE

The aeroacoustic interface is a large data set whose generation will tax even today's computational facilities. The unsteady boundary layer and wake must be given as a function of time. Assuming that the wake surface is given by two surface parameters, the spanwise origin of a vorticity filament and a wake age variable, then about 10^5 data elements will be needed to describe the wake motion in time. Considering ten aerodynamic variables such as density, position, velocity, and vorticity, this interface has around 10^6 data elements.


AERODYNAMIC MODULES
10^3 WAKE POSITIONS
10^2 TIMES
10 FUNCTIONS
POSITION
DENSITY
VELOCITY
VORTICITY
10^6 DATA ELEMENTS
SUBCOMPONENT NOISE PREDICTION

CONCLUSIONS

Because of the limitations of empirical methods by the number of parameters and the limitations of analytical methods by available computational facilities, rotorcraft noise prediction requires a blended empirical and analytical approach on the component or subcomponent level. Available analytical methods must be extended to allow for stochastic effects in all variables and a better understanding of the covariances of signals from different subcomponents must be developed in order to sum the subcomponent noise. An improved definition of the interface between aerodynamics and acoustics is needed. The blade surface may not be adequate due to quadrupole source errors so that the boundary layer/wake surface is suggested as an alternative aeroacoustic interface.

- **ROTORCRAFT NOISE PREDICTION REQUIRES A BLENDED EMPIRICAL AND ANALYTICAL APPROACH ON THE COMPONENT OR SUBCOMPONENT LEVEL**
- **A BETTER UNDERSTANDING OF THE COVARIANCES OF SIGNALS FROM DIFFERENT SUBCOMPONENTS IS NEEDED FOR ANALYTICAL PREDICTION OF COMPONENT NOISE**
- **AN IMPROVED DEFINITION OF THE INTERFACE BETWEEN AERODYNAMICS AND ACOUSTICS IS NEEDED. THE BOUNDARY LAYER/WAKE SURFACE IS SUGGESTED AS AN ALTERNATIVE TO THE BLADE SURFACE**

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ROTOR NOISE PREDICTION TECHNOLOGY - THEORETICAL APPROACH

F. Farassat
NASA Langley Research Center
Hampton, Virginia

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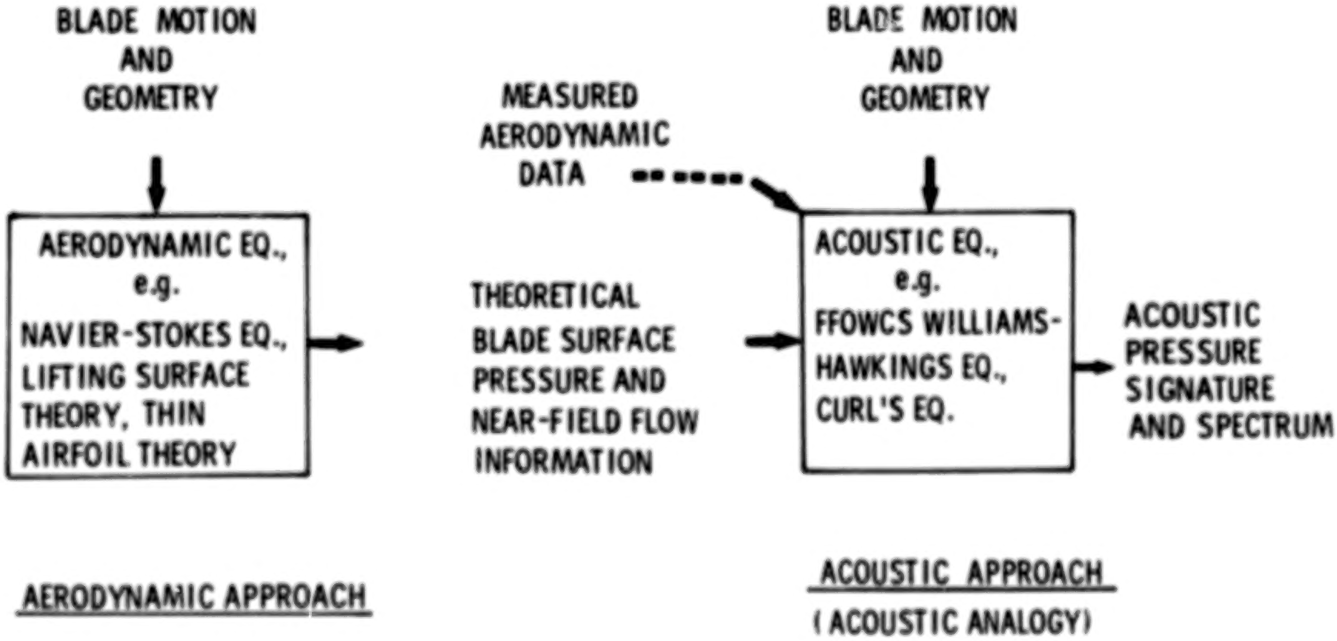
THE ACOUSTIC AND AERODYNAMIC APPROACHES OF PROBLEM SOLVING

Helicopter rotors are perhaps the most complicated noise generators among rotating blade machinery. Many mechanisms have been found to generate noise in the operation of helicopter rotors. The relative importance of the noise generated by these mechanisms varies with both the operating conditions and the observer position. The main or the tail rotor can be the dominant source of noise depending on the range of frequencies and the observer position. Significant advances in noise prediction have been made in recent years, but more work needs to be done. The most successful and general theoretical method for treating the acoustics of helicopter noise is through the use of the acoustic analogy. The figure below explains the basic idea behind acoustic analogy. In this figure, the approach of solving aerodynamic problems is also shown. It appears that to study nonlinear acoustic effects one has to use the aerodynamic approach. The results of acoustic analogy depend greatly on the input data. A significant problem is to supply the aerodynamic input data either by a theoretical or experimental technique.

THE DIFFERENT APPROACHES OF SOLVING AERODYNAMIC AND ACOUSTIC PROBLEMS


BLADE MOTION AND GEOMETRY
AERODYNAMIC EQ., e.g. NAVIER-STOKES EQ., LIFTING SURFACE THEORY, THIN AIRFOIL THEORY
MEASURED AERODYNAMIC DATA
THEORETICAL BLADE SURFACE PRESSURE AND NEAR-FIELD FLOW INFORMATION
BLADE MOTION AND GEOMETRY
ACOUSTIC EQ., e.g. FFOWCS WILLIAMS-HAWKINGS EQ., CURL'S EQ.
ACOUSTIC PRESSURE SIGNATURE AND SPECTRUM
AERODYNAMIC APPROACH
ACOUSTIC APPROACH
(ACOUSTIC ANALOGY)

THE GOVERNING DIFFERENTIAL EQUATION

The most general governing differential equation for acoustics of helicopter rotors and propellers is the Ffowcs Williams-Hawkings equation (FW-H Eq.). This equation is derived by the use of acoustic analogy. It can be shown that one can derive most, if not all, of previous linear acoustic formulas on the noise generated by moving bodies. Many linear aerodynamic results can also be derived from FW-H equation. The left side of FW-H equation is the linear wave operator acting on $c^2(\rho-\rho_0)$ which can be interpreted as acoustic pressure when $\rho-\rho_0 << \rho_0$. There are three source terms on the right of FW-H Eq. Two are surface sources which depend on the forces acting on the fluid and the normal velocity of the blade surface, respectively. The third term is a volume source which accounts for turbulence noise and nonlinear effects. The FW-H equation has received a great deal of attention since the mid 70's. It appears that, except perhaps for the study of nonlinear effects, this equation is ideally suited for noise calculation.

CLASSIFICATION OF NOISE GENERATING MECHANISMS ACCORDING TO THEIR CONTRIBUTION TO THE SOURCE TERMS OF FW-H EQUATION

$$\square^2 p' = \frac{\partial}{\partial t}\left[\rho_0 v_n \left|\nabla f\right| \delta(f)\right]$$

- THICKNESS NOISE
- NOISE DUE TO BLADE VIBRATION

$$-\frac{\partial}{\partial x_i}\left[\ell_i \left|\nabla f\right| \delta(f)\right]$$

- STEADY AND PERIODIC BLADE LOADING NOISE
- UNSTEADY BLADE LOADING NOISE
  - BLADE/VORTEX INTERACTION
  - BLADE/TURBULENCE INTERACTION
  - BOUNDARY LAYER FLUCTUATIONS
  - VORTEX SHEDDING
  - TRAILING EDGE FLUCTUATIONS

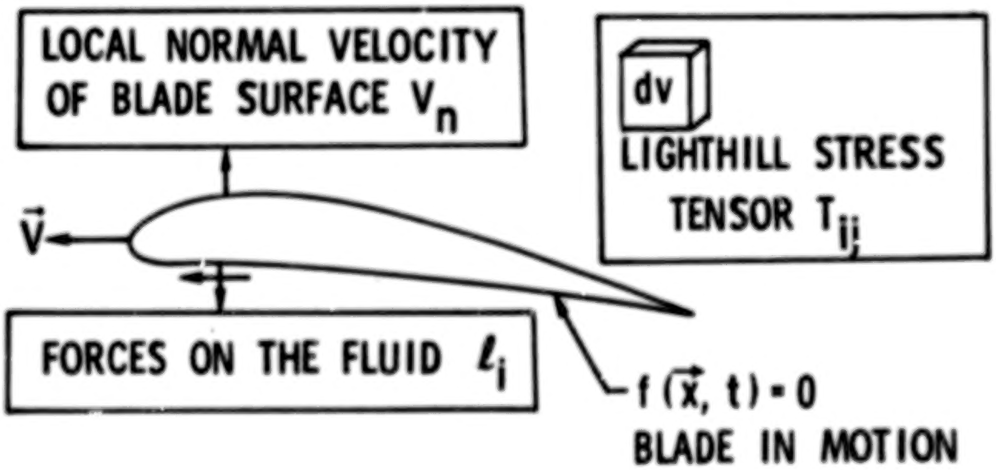
$$+\frac{\partial^2}{\partial x_i \partial x_j}\left[T_{ij} H(f)\right]$$

- NONLINEAR EFFECTS
- SHOCK NOISE

THE SOURCES IN FW-H EQUATION AND THEIR RELATION TO NOISE GENERATION MECHANISMS

From the point of view of acoustic analogy, there are basically two kinds of surface sources - those depending on surface normal velocity distribution and those which depend on surface forces acting on the medium. Most of the mechanisms of noise generation of helicopter rotors can be grouped under these two categories as shown below.

THE SOURCES APPEARING IN FW-H EQUATION


LOCAL NORMAL VELOCITY OF BLADE SURFACE V_n
dv
LIGHTHILL STRESS TENSOR T_{ij}
$\vec{V}$
FORCES ON THE FLUID ℓ_i
$f(\vec{x}, t) = 0$
BLADE IN MOTION

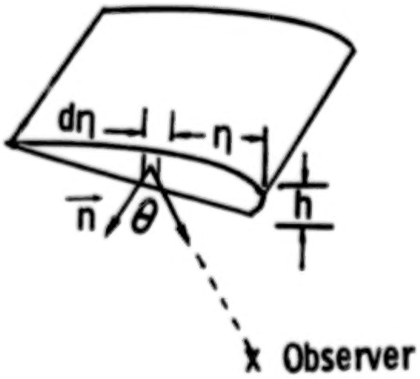
SOLUTION OF FW-H EQUATION

The solution of the FW-H equation can be written in many forms. The solution in time domain which is programed for use on a computer at Langley is shown below. Note that this formulation is for discrete frequency noise calculation. For broadband noise calculation, a solution in frequency domain is required. The solution, as programed, included the thickness and loading noise components. The line integral over the tip chord is the contribution of the pressure distribution over the edge of the blade tip. It is found to contribute to the blade noise in some cases of interest. The tip noise is in phase with thickness noise. The program is being modified to include a portion of the quadrupole term which requires only the knowledge of the T_{ij} term on the blade only. This is done to improve prediction at transonic speed.

THE SOLUTION OF FW-H EQUATION PROGRAMMED AT LANGLEY

$$4\pi p'(\vec{x},t) = \frac{1}{c}\frac{\partial}{\partial t}\int_{f=0}\left[\frac{\rho_0 c v_n + \ell_r}{r(1-M_r)}\right]_{ret} ds + \int_{f=0}\left[\frac{\ell_r}{r^2(1-M_r)}\right]_{ret} ds$$

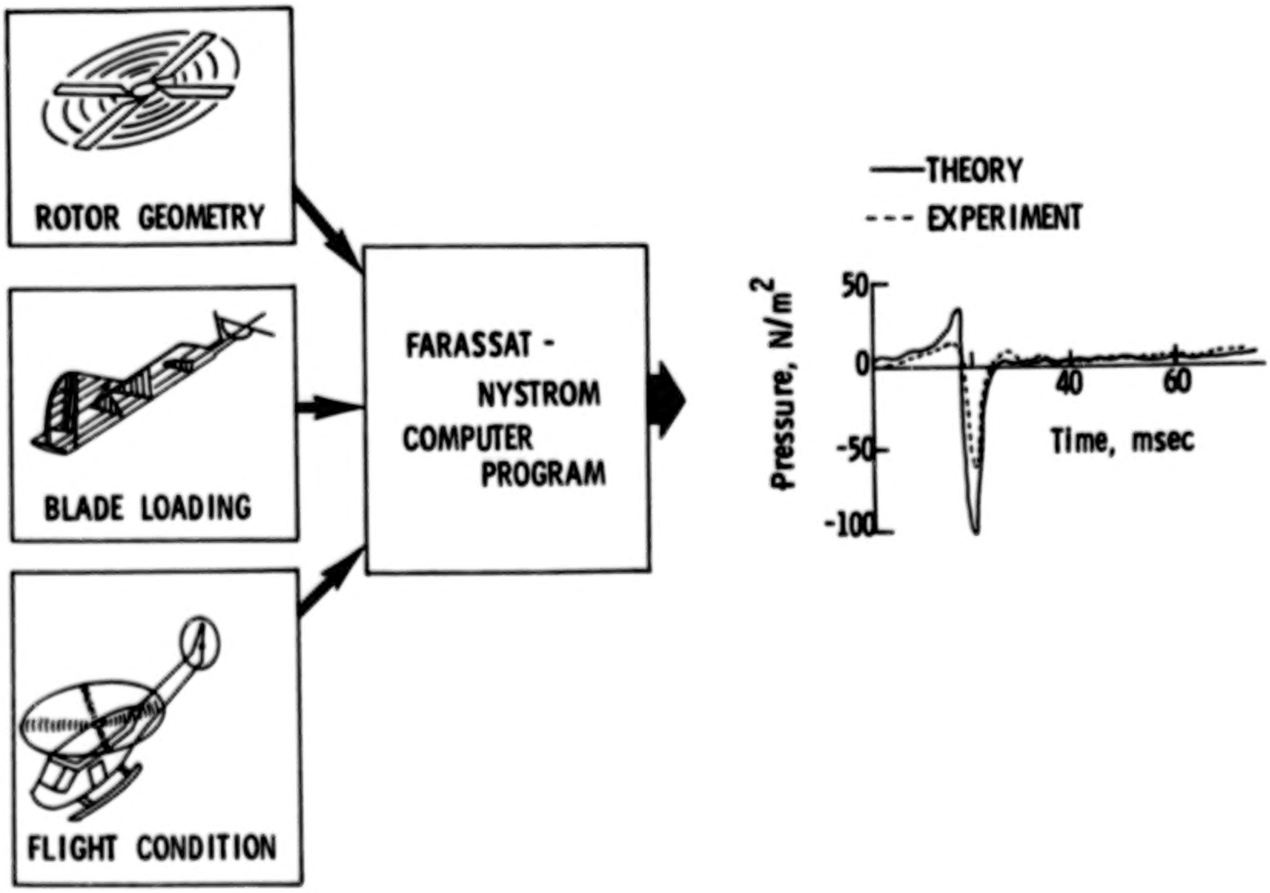
$$+\frac{1}{c}\frac{\partial}{\partial t}\int_{\substack{\text{Tip}\\ \text{Chord}}}\left[\frac{h\,p\cos\theta}{r(1-M_r)}\right]_{ret} d\eta + \int_{\substack{\text{Tip}\\ \text{Chord}}}\left[\frac{hp\cos\theta}{r^2(1-M_r)}\right]_{ret} d\eta$$



THE LANGLEY HELICOPTER ROTOR NOISE PROGRAM

The figure below shows the elements of the Langley program. This is in an experimental stage. It will be updated to include broadband noise at a later time. The program is written to model precisely the geometry and motion of the blade. For example, rotor angle of attack, coning and flapping angle changes, and azimuthal blade pitch variation are included in the noise calculation. This was done to reduce a source of error in modeling of rotor noise whose importance is difficult to assess. The observer can be stationary (fixed to the ground) or moving with the helicopter.

HELICOPTER NOISE PREDICTION


ROTOR GEOMETRY
BLADE LOADING
FLIGHT CONDITION
FARASSAT - NYSTROM COMPUTER PROGRAM
—— THEORY
--- EXPERIMENT
Pressure, N/m^2
50
0
-50
-100
40
60
Time, msec

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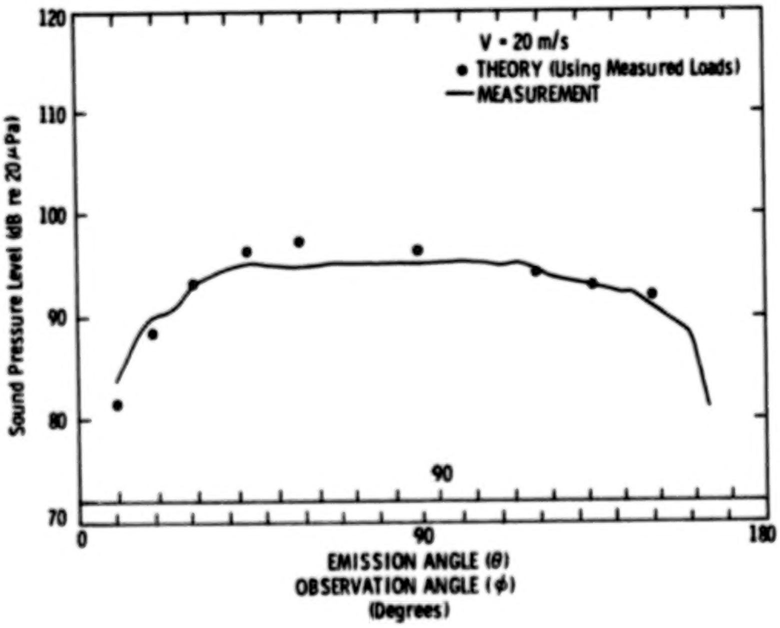
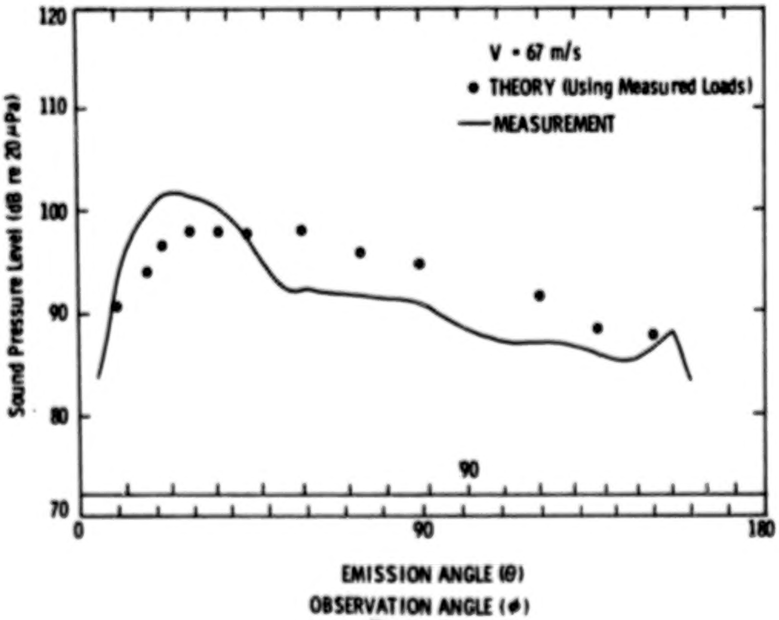
EXAMPLES OF APPLICATION

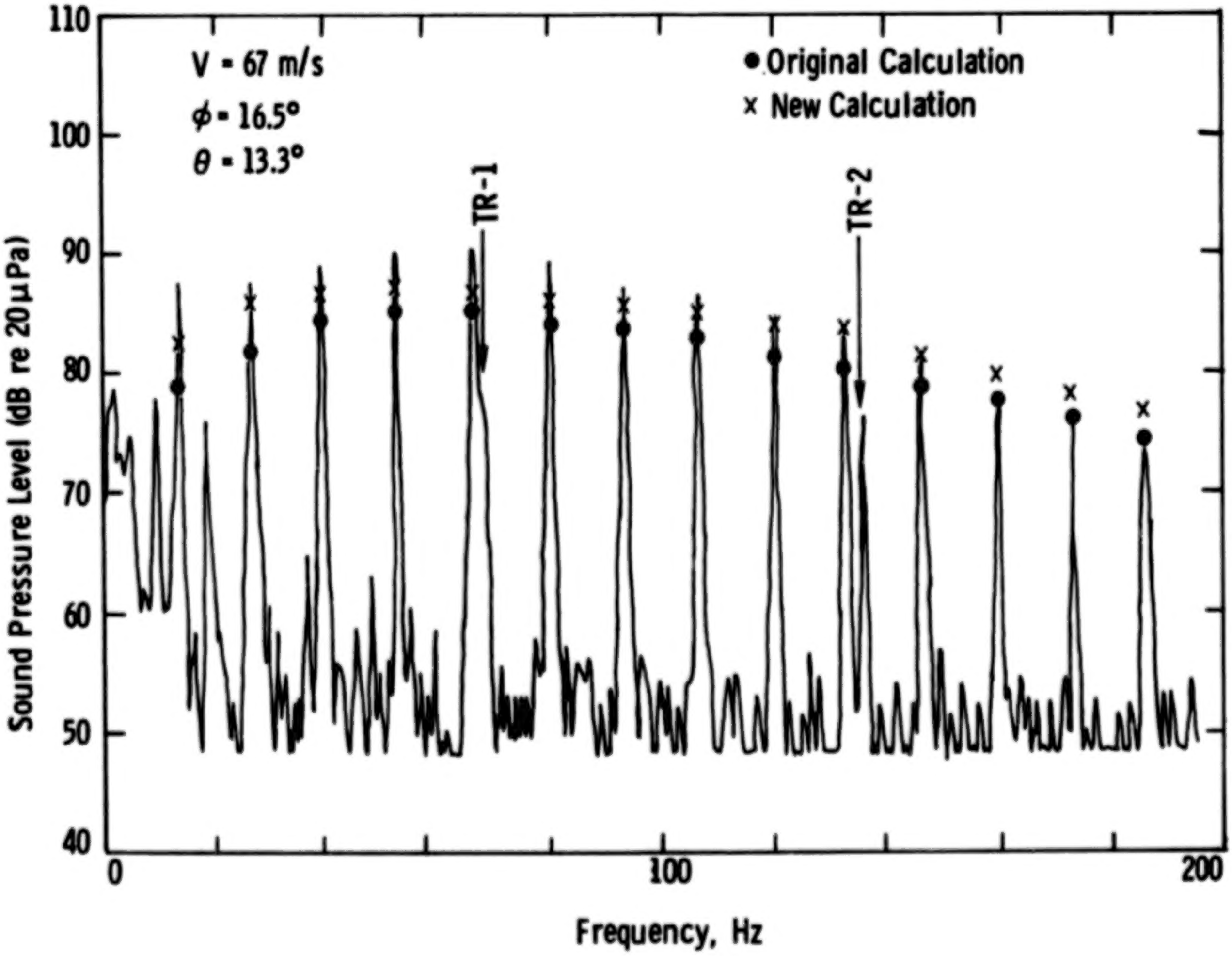
The two examples below show sample calculations for a two-bladed helicopter. These calculations were performed at BBN by G. P. Succi. The overall acoustic pressure spectra are shown at two different flight speeds. It must be mentioned that the mean helicopter forward speed and measured loads were used in these calculations. However, there were not enough pressure transducers on the instrumented blade to get a fine resolution of pressure distribution. Some assumptions had to be used to fill in the gaps in aerodynamic data. It is seen that at the lower speed the agreement between measured and calculated data is good. However, for the higher speed there is disagreement at some observation angles. The next slide will show improvement in the predicted spectrum when more accurate forward speed and a better model of blade tip region surface pressure are used.





THE EFFECT OF IMPROVED INPUT DATA

The figure below shows improvement in agreement of measured and predicted acoustic spectra when better input data were used. In the new calculations four changes were made as follows. The instantaneous helicopter velocity at emission time was used. The actual blade surface pressure (rather than the pressure difference between the upper and lower surfaces as in the original calculation) was used. Also, a better model of upper and lower surface pressure near the blade tip and an approximate tip side edge suction pressure were employed. The underestimation of the first few harmonics can be attributed to the nonlinear effects and shock noise.



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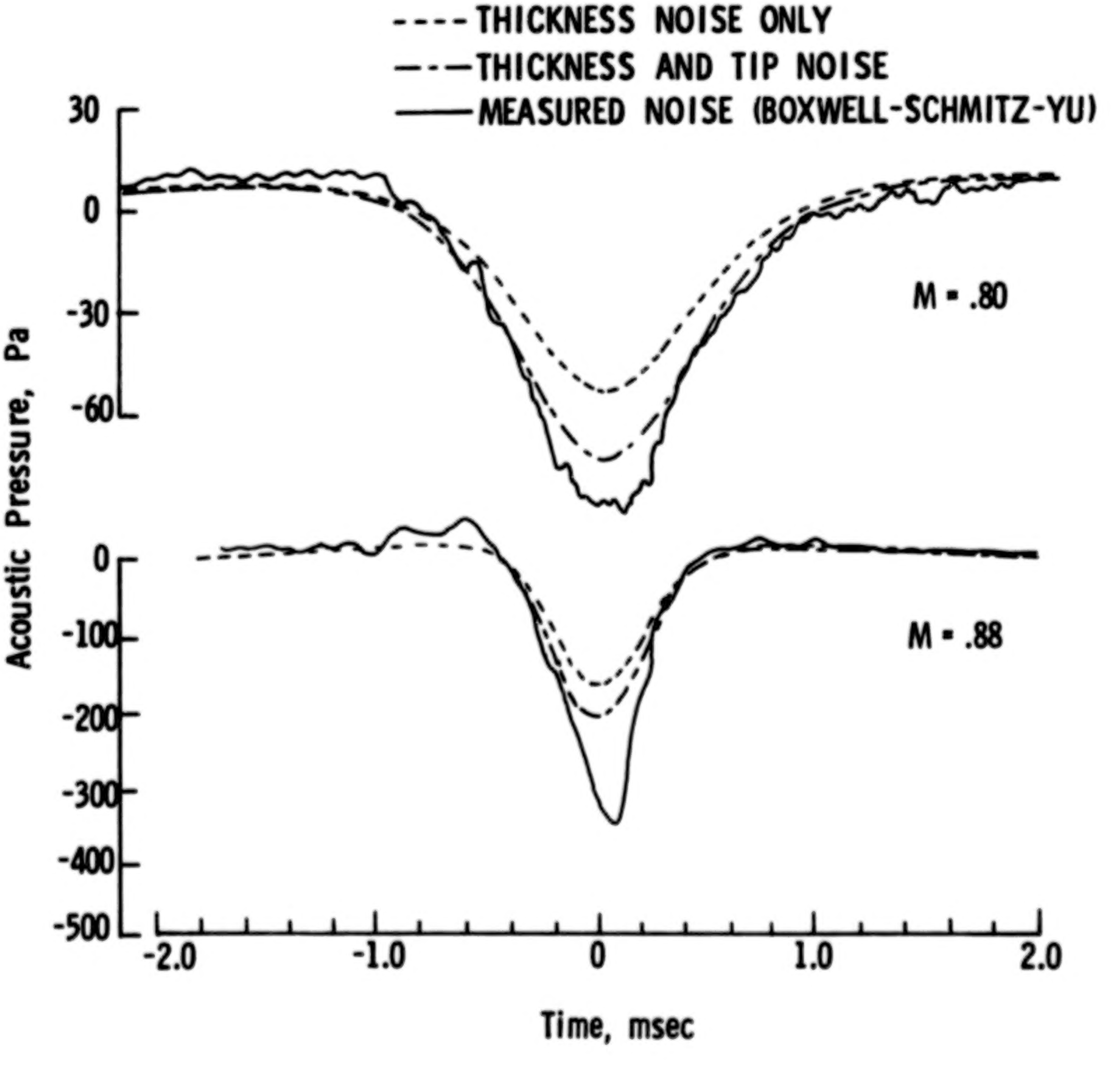
TIP NOISE OF A MODEL ROTOR

The figure below shows two examples where improvement in input data has resulted in better agreement of prediction with measured noise data. As the result of investigation of a thickness noise formulation of M. P. Isom, it was suspected that the suction force at the tip of thick helicopter blades can generate noise in phase with thickness noise. This suction force was estimated and used in the calculation of rotor noise for which experimental data were available. It is seen that in both cases shown below the agreement of prediction with experimental results is improved.

THE EFFECT OF TIP NOISE ON THE ACOUSTIC PRESSURE SIGNATURE OF A MODEL HELICOPTER ROTOR



CONCLUDING REMARKS

The prediction of the noise of helicopter rotors involves both aerodynamics and acoustics. The approach of the acoustic analogy has shifted the burden to the aerodynamicist. As long as the surface sources are important, acoustic analogy seems the right approach. For the study of nonlinear effects, acoustic analogy may be too cumbersome and inefficient to apply. The aerodynamic problems related to noise generation are of the type which have not been worked out by helicopter aerodynamicists. There is also a shortage of good aerodynamic and acoustic data for validation of prediction techniques. It is hoped that efforts in solving aerodynamic problems and formation of a data base for both aerodynamic and acoustic data will result in better rotor noise prediction capability.

CURRENT STATE OF HELICOPTER ROTOR NOISE PREDICTION TECHNOLOGY

- COMPARISON OF PREDICTIONS WITH THE LIMITED AVAILABLE ACOUSTIC MEASUREMENTS HAVE BEEN ENCOURAGING.
- USING PRESENT DAY COMPUTERS, IT IS POSSIBLE TO SPECIFY AS INPUT REALISTIC DESCRIPTIONS OF ROTOR GEOMETRY, KINEMATICS, AND SURFACE PRESSURE FOR NOISE CALCULATIONS.
- ALTHOUGH THE FORMULATIONS BASED ON THE ACOUSTIC ANALOGY ARE NUMEROUS AND IN ADVANCED STAGE OF DEVELOPMENT, ONE SHOULD REMEMBER THAT THE METHODOLOGY FOR OBTAINING THE REQUIRED THEORETICAL OR MEASURED AERODYNAMIC INPUT DATA IS INCOMPLETE AND NEEDS FURTHER DEVELOPMENT.
- THERE IS A SHORTAGE OF ACCURATE AERODYNAMIC INPUT DATA AND MEASURED ACOUSTIC DATA REQUIRED FOR VALIDATING ADVANCED PREDICTION METHODS.
- AERODYNAMIC PROBLEMS DIRECTLY RELATED TO ROTOR NOISE GENERATION (e.g. BLADE/ VORTEX, BLADE/TURBULENCE INTERACTION) MUST BE ATTACKED AND SOLVED. TESTS TO VALIDATE THEORETICAL RESULTS MUST BE PERFORMED.

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EXPERIMENT VERSUS THEORY

FREDERIC H. SCHMITZ
YUNG H. YU
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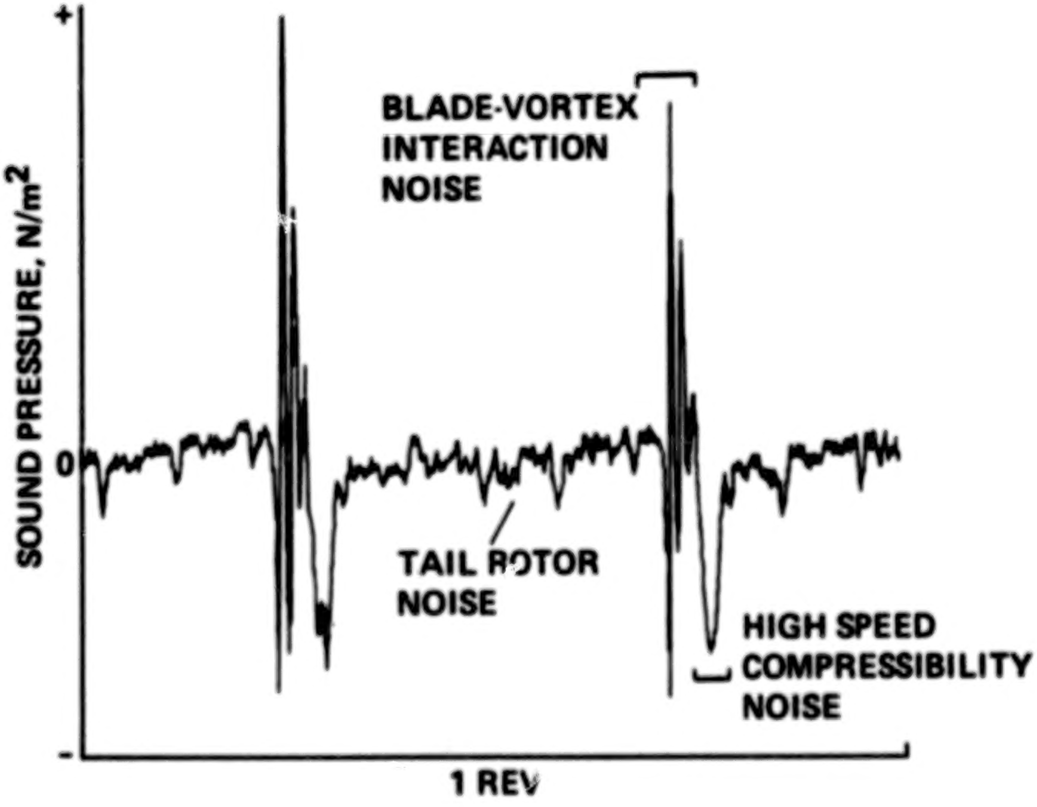
IMPORTANT AERODYNAMICALLY GENERATED NOISE SOURCES

- High Speed Compressibility Noise
- Blade-Vortex Interaction Noise



HIGH SPEED COMPRESSIBILITY NOISE

(1) Experiment:

- Anechoic Hover Chamber at Aeromechanics Lab
- 7' x 10' Wind Tunnel at Aeromechanics Lab
- YO-3A Technique for Full Scale Test
- DNW Anechoic Wind Tunnel, Holland

(2) Theory:

Ffowcs Williams and Hawkings Formula (1969)

- Monopole & Dipole
- Quadrupole

$$4\pi a_o^2 \rho'(\vec{X},t) = \frac{\partial}{\partial t}\iint_S \left[\frac{\rho_o U_n}{r\,|1-M_r|}\right] dS(\vec{y})$$

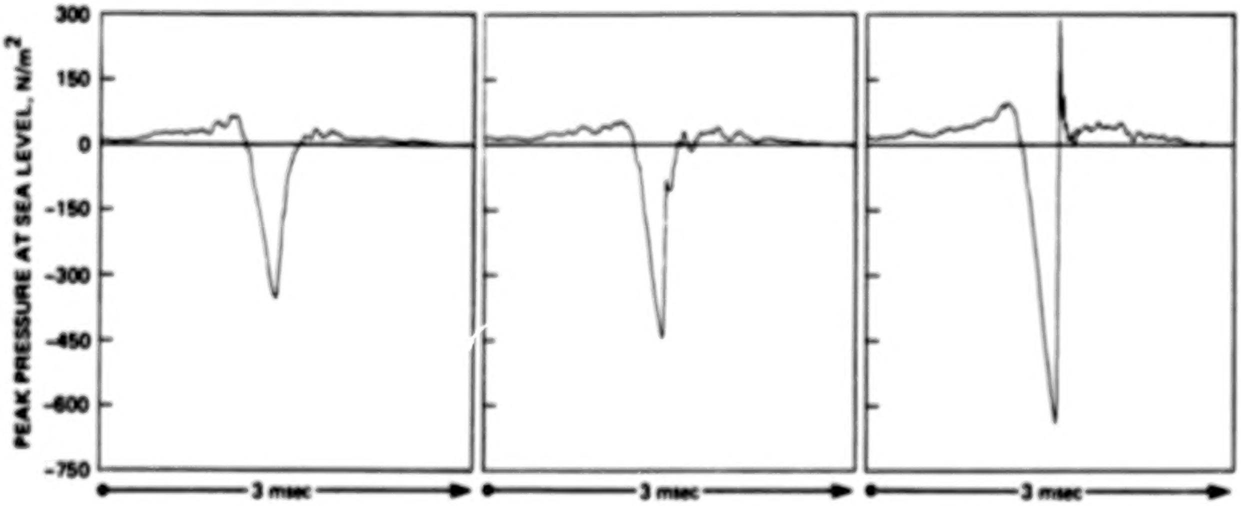
$$-\frac{\partial}{\partial x_i}\iint_S \left[\frac{P_{ij} n_j}{r\,|1-M_r|}\right] dS(\vec{y})$$

$$+\frac{\partial^2}{\partial x_i\,\partial x_j}\iint_V \left[\frac{T_{ij}}{r\,|1-M_r|}\right] dV(\vec{y})$$

$$T_{ij} = \rho u_i u_j + P_{ij} - a_o^2 \rho \delta_{ij}$$

HIGH SPEED COMPRESSIBILITY NOISE (Cont.)

- Waveform transition
  - symmetric shape for M =0.88 and below
  - saw-tooth shape for M =0.9 and above
    for NACA0012
    Aspect ratio = 13.7
    Hovering case



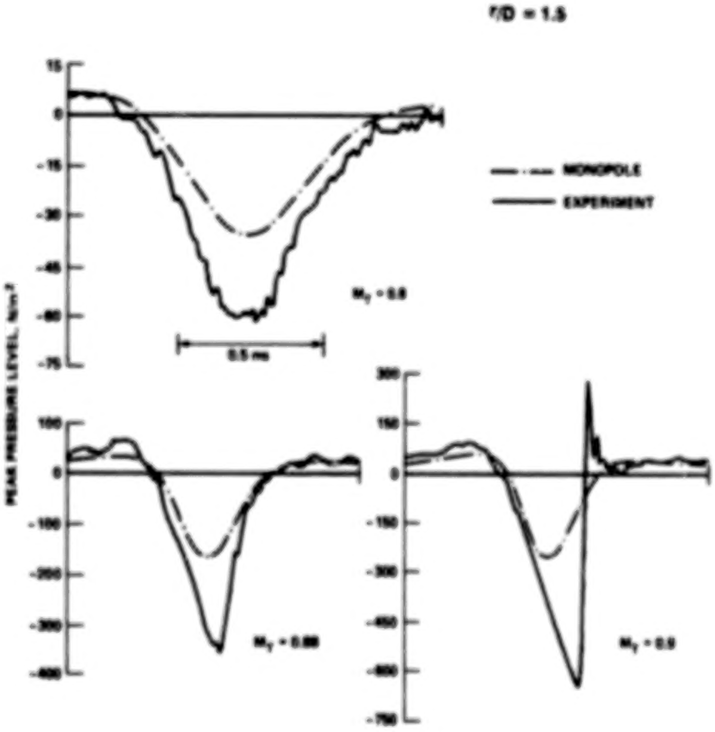
M_T = 0.88
(a)

M_T = 0.89
(b)

M_T = 0.90
(c)

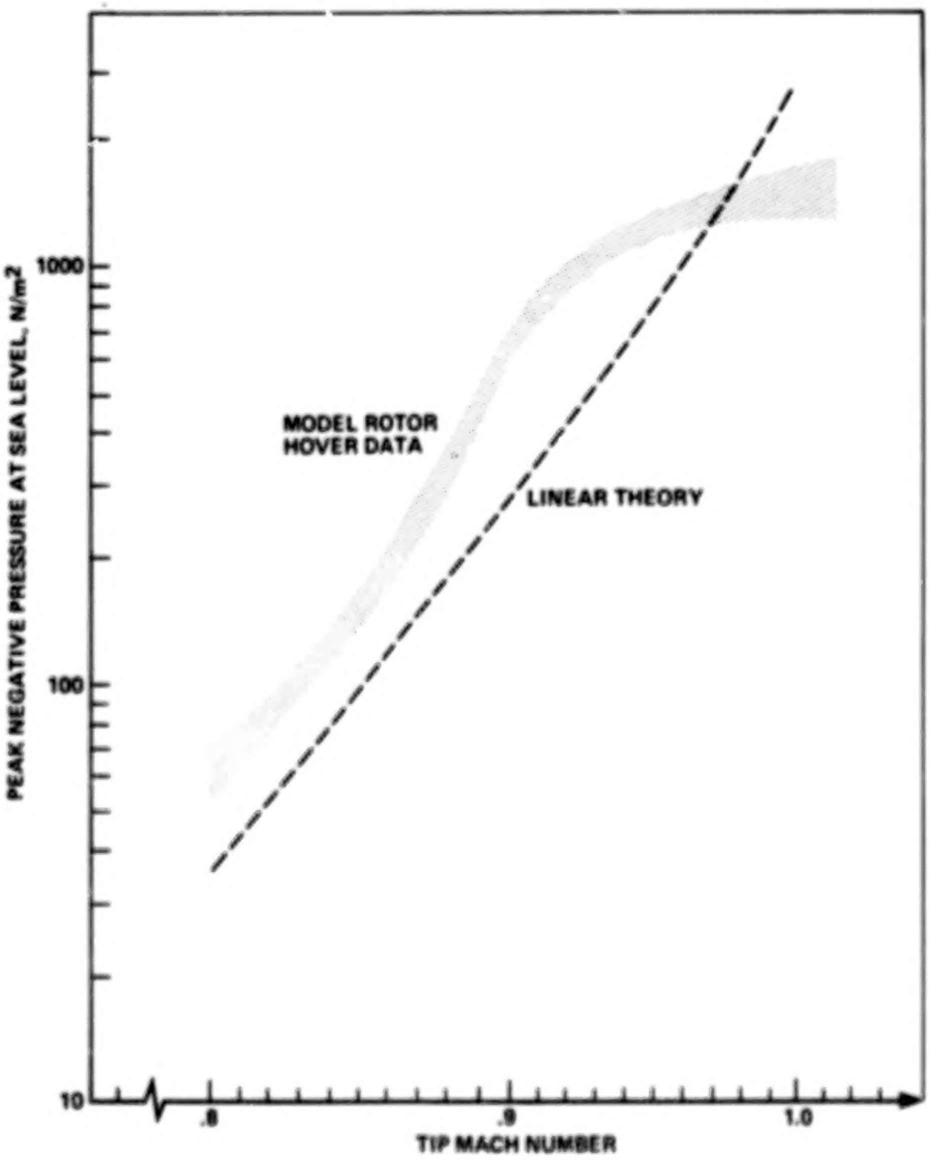
HIGH SPEED COMPRESSIBILITY NOISE (Cont.)

- Comparison of Experiment and Linear Theory
  - underpredict amplitude
  - wrong waveform shape at M =0.9 and above



HIGH SPEED COMPRESSIBILITY NOISE (Cont.)

- Comparison of experiment and linear theory
  - amplitude only



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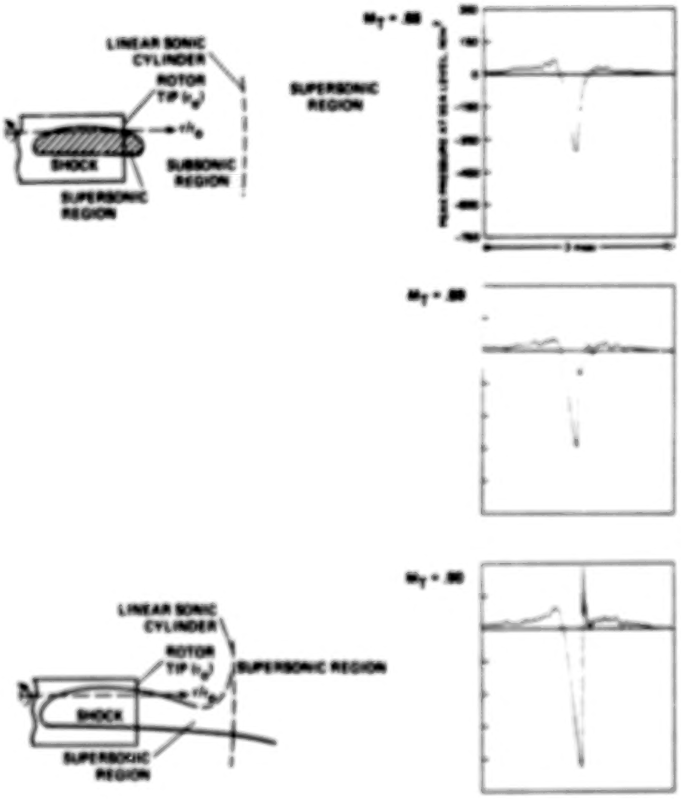
HIGH SPEED COMPRESSIBILITY NOISE (Cont.)

NONLINEAR TERM



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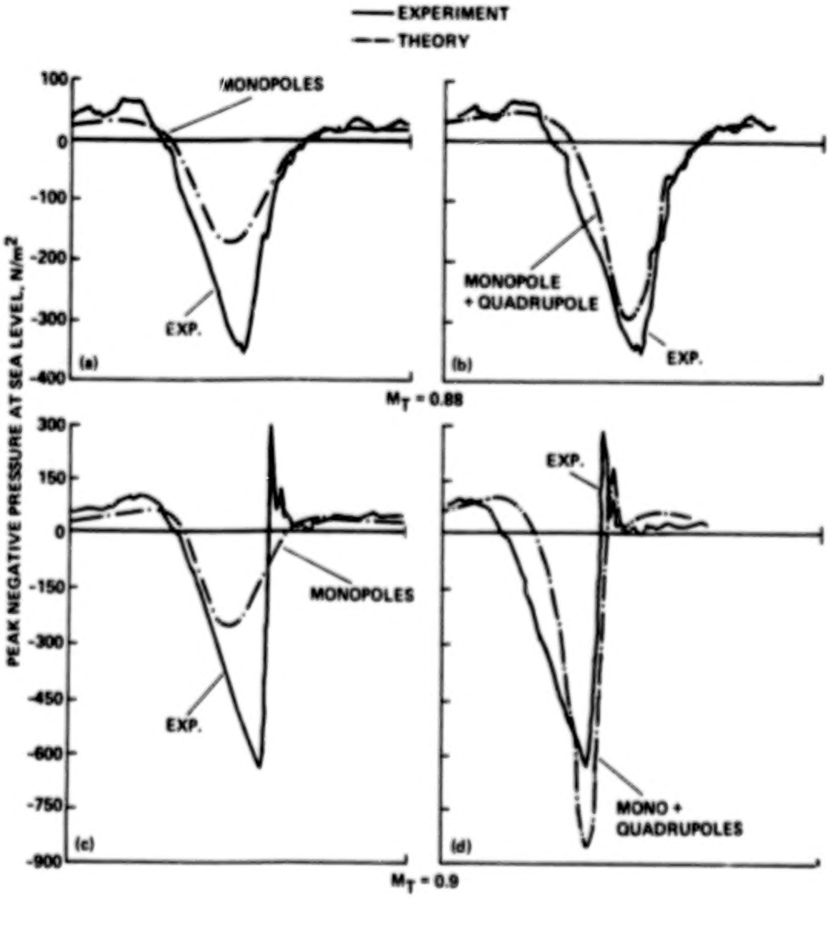
HIGH SPEED COMPRESSIBILITY NOISE (Cont.)

- Comparison of theory (linear, nonlinear terms) and experiment
  - linear theory ---- monopole only
  - nonlinear theory ---- monopole and quadrupole
- Nonlinear theory
  - improve the amplitude
  - generate the right waveform shape at the right Mach number



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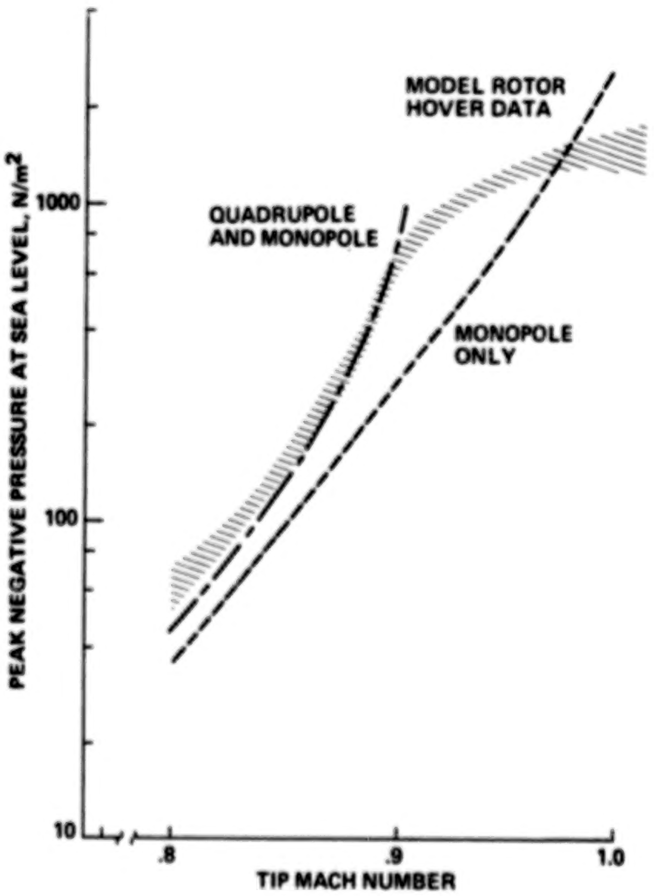
HIGH SPEED COMPRESSIBILITY NOISE (Cont.)

- Comparison of theory and experiment
  - amplitude only

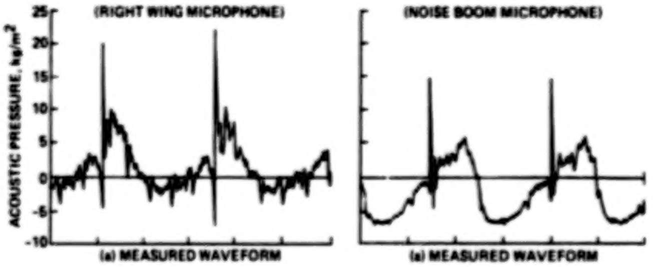


BLADE-VORTEX INTERACTION NOISE

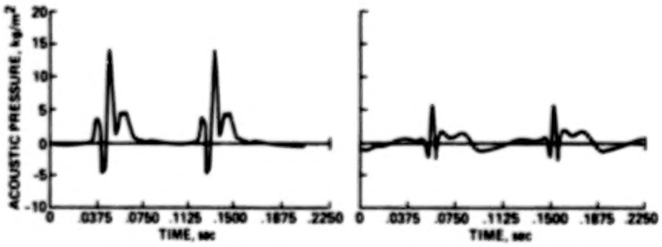
- Simultaneous measurements of blade surface pressure and acoustics
  - Army/Bell Helicopter joint program
    AH-1G Operational Load Survey (OLS)
- Monopole and dipole terms of Ffowcs Williams and Hawkings formula

BLADE-VORTEX INTERACTION NOISE (Cont.)

Theory and Experiment Comparison



(a) MEASURED WAVEFORM (a) MEASURED WAVEFORM



(b) COMPUTED WAVEFORM (b) COMPUTED WAVEFORM

CONCLUDING REMARKS

- Find important noise generating mechanisms
- Experiment - carefully controlled experiments
- Theory - Transonic Aerodynamics should be considered

FOR HIGH SPEED COMPRESSIBILITY NOISE

- linear and nonlinear theory agree very well with experiment data in terms of amplitude and waveform
- linear acoustic code available (computer time, CDC7600 less than a minute)
- nonlinear acoustic code will be available soon along with a small disturbance transonic code (steady and unsteady)

FOR BLADE-VORTEX INTERACTION NOISE

- High quality information over interaction area between the acoustic planform and blade-vortex interaction lines
- nonlinear term may be needed

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AERODYNAMIC PREDICTION I

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AERODYNAMIC PREDICTION FOR ROTORCRAFT NOISE CALCULATIONS

This presentation gives a review of the current capabilities and future requirements of the aerodynamic prediction methodology needed for rotor noise calculations. The major aerodynamic topics reviewed are airloads and wakes, dynamic stall, compressible flow, and random loads.

The aerodynamics/acoustics interface appears well defined when the problem (noise) or measurements are considered but is becoming less so for the calculation task. The classical approach involving a distinct separation of aerodynamic sources and acoustic propagation is the basis of most rotor noise calculations. Some problems are however more appropriately treated as a single fluid dynamic phenomenon. Possible examples are when shocks extend into the acoustic field or when noise influences blade random aerodynamic loads.

Aeroacoustics of rotors do involve unique features not encountered in other aerodynamic problems of rotors. First, acoustics problems ultimately require only an order of magnitude estimate of the sound pressure field. Consequently there are cases where remarkably good noise predictions have been obtained from very simple aerodynamic analyses. Second, aeroacoustics introduces an interest in the higher frequency aerodynamics. The deterministic aerodynamic phenomena are still low frequency; however, the high frequency noise comes from the acoustic propagation in this case. Third, there are aerodynamic phenomena on rotors that are only of interest due to the noise they produce. An example is the random aerodynamic pressures on the blades.

- REVIEW OF CURRENT CAPABILITIES AND FUTURE REQUIREMENTS, AS RELATED TO ROTOR NOISE PREDICTION
  - ---- AIRLOADS AND WAKES
  - ---- DYNAMIC STALL
  - ---- COMPRESSIBLE FLOW
  - ---- RANDOM LOADS
- AERODYNAMICS/ACOUSTICS INTERFACE
  - ---- DISTINCTION BETWEEN AERODYNAMICS AND ACOUSTICS IN CALCULATIONS
  - ---- UNIQUE REQUIREMENTS OF AERODYNAMICS FOR NOISE PREDICTIONS

AIRLOADS AND WAKES

The following figures show examples of rotor blade airloads and wake velocity predictions. With work it should be possible to do better now (these results are all several years old); it is also easy to do worse.

The level of accuracy possible in the airloads calculations implies for the noise calculations some directivity shifts, perhaps 1 or 2 dB error in amplitudes, or some shifts in the operating conditions where particular phenomena occur. Such errors would not be of concern in terms of predicting how loud the helicopter is. But this level of accuracy does complicate the correlation task since it means measurements and predictions can not be compared at one point in space/time and at exactly the same operating condition. Either the measurements or the predictions, or both, must cover a range of positions and operating conditions. The sensitivity of the noise to changes in position or operating condition becomes even more important.

The rotor airloads are used to predict the rotational noise. It should be noted that the extraction of the deterministic part of the noise is not the same for the measurements and predictions. The noise signal is averaged in the time domain for the measurements; while for the predictions a periodic airloads solution is obtained. For linear problems the results of these two operations would be identical. But the rotor airloads calculation involves a geometric nonlinearity in the influence of the wake position of the loading. There is no quantitative basis to be concerned about this difference now. If it is necessary to make the calculation match the measurement, it would greatly complicate the prediction.

The prediction of rotor airloads is presently based on empirical models for a number of key aerodynamic phenomena. The aerodynamic prediction will not be truly adequate for noise predictions until a first principles solution is possible.

These results show that if the problem is simple enough a good calculation of the rotational noise is possible. Of course, the real case is not this simple; but it does emphasize that the basic problem is the rotor wake.

- EXAMPLES OF CAPABILITY TO PREDICT ROTOR AIRLOADS AND WAKE VELOCITIES

  ---- THE BASIC PHENOMENA ARE BEING CALCULATED, OR AT LEAST MODELLED

- IMPLICATIONS FOR ROTATIONAL NOISE CALCULATIONS

  ---- LEVEL OF ACCURACY OF AIRLOADS CALCULATIONS STILL REQUIRES CONSIDERATION OF SENSITIVITY TO POSITION AND OPERATING STATE NOISE CALCULATIONS

  ---- EXTRACTION OF DETERMINISTIC NOISE SIGNAL IS NOT THE SAME FOR MEASUREMENTS AND CALCULATIONS, DUE TO NONLINEAR INFLUENCE OF WAKE POSITION ON BLADE LOADS

  ---- AERODYNAMIC PREDICTIONS ARE NOT YET OBTAINED FROM FIRST PRINCIPLES ANALYSES

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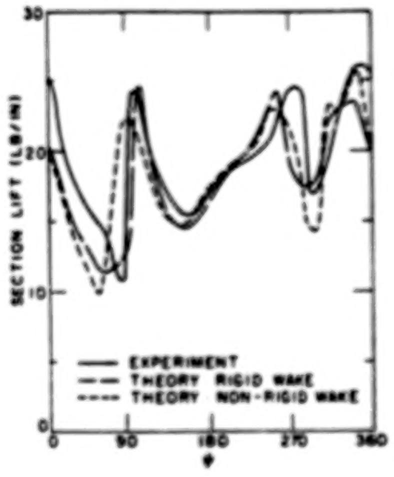
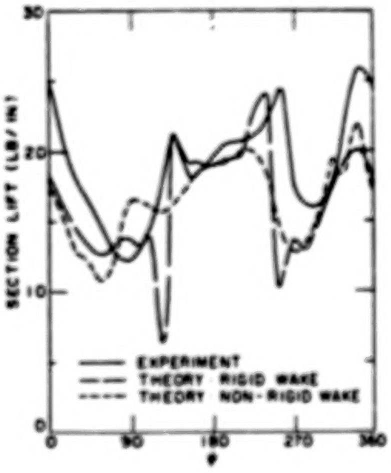
AIRLOADS AND WAKES - Continued

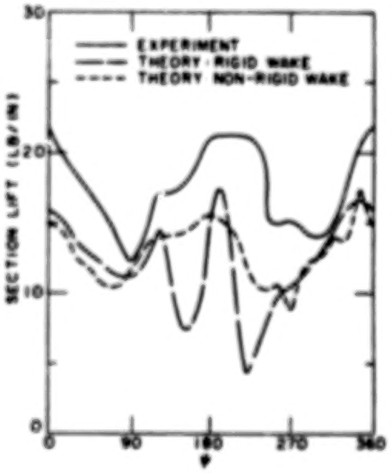
Example of rotor airloads calculations (Johnson, 1971):



$\mu = 0.15;\ \theta^{\circ} = 12.5^{\circ};$
$r = 0.95$



$\mu = 0.15;\ \theta^{\circ} = 12.5^{\circ};$
$r = 0.85$

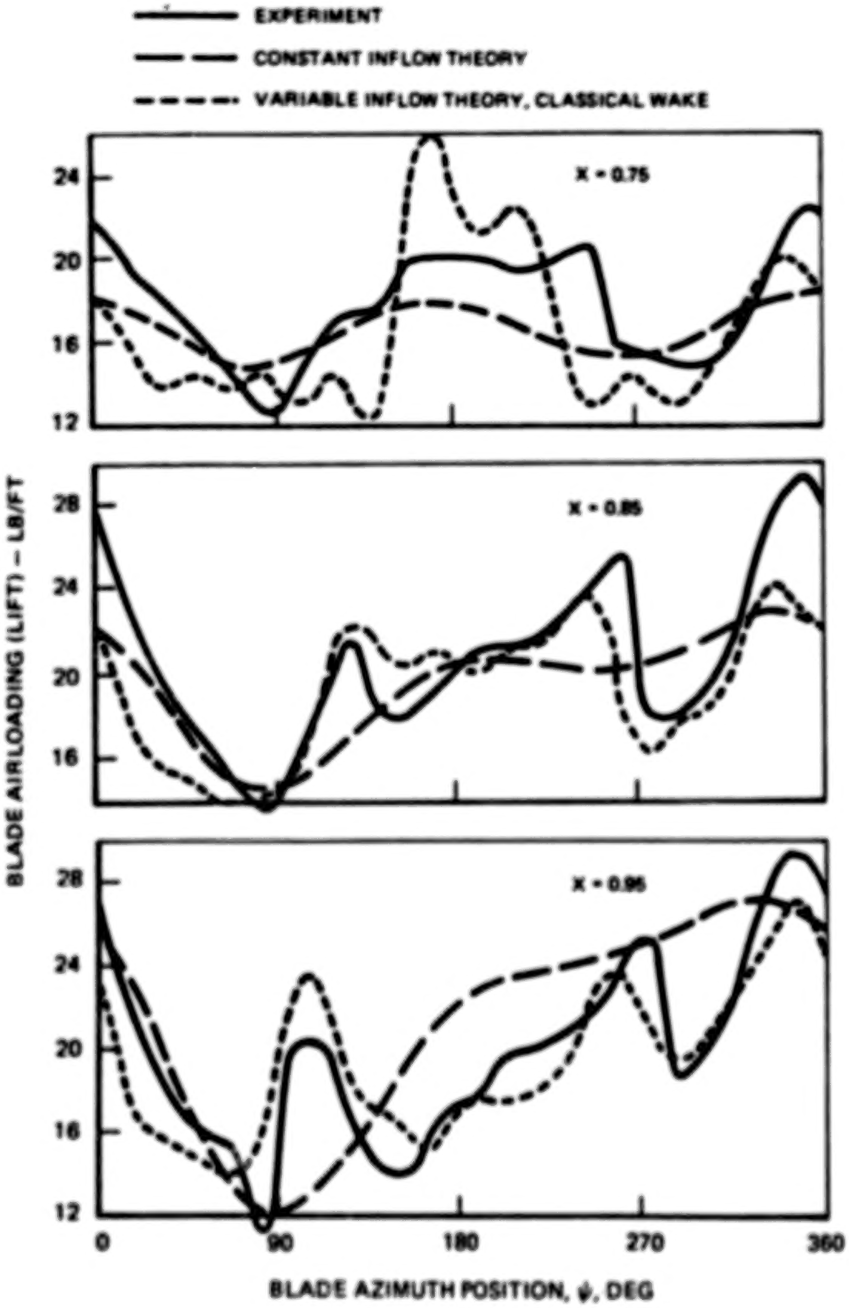


$\mu = 0.15;\ \theta^{\circ} = 12.5^{\circ};$
$r = 0.75$

Section lift calculated with lifting-surface theory vs experimental results.

AIRLOADS AND WAKES - Continued

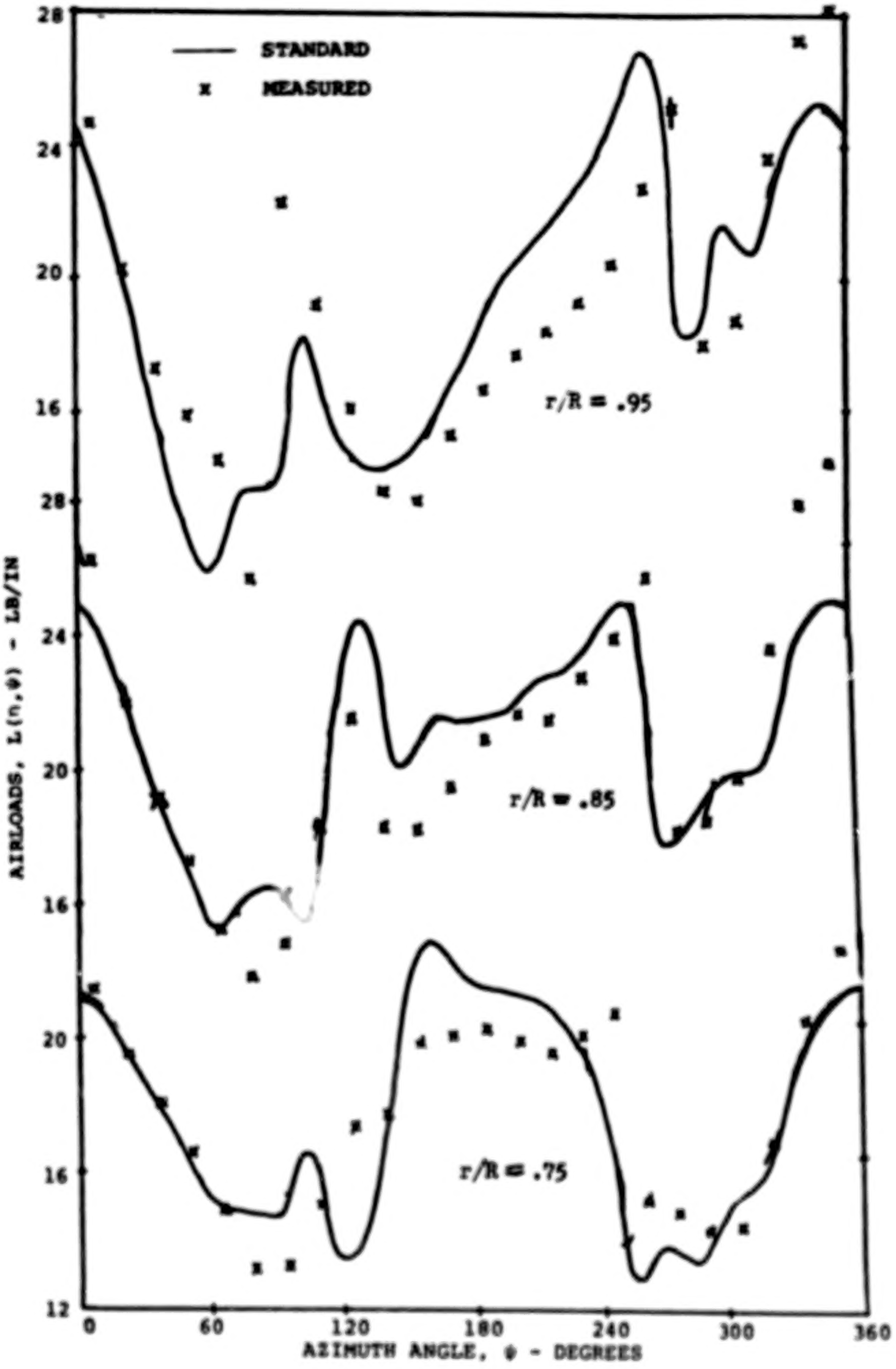
Example of rotor airloads calculations (Landgrebe and Egolf, 1976a):



Comparison of analytical and experimental blade airloads.
H-34, 70 knots.

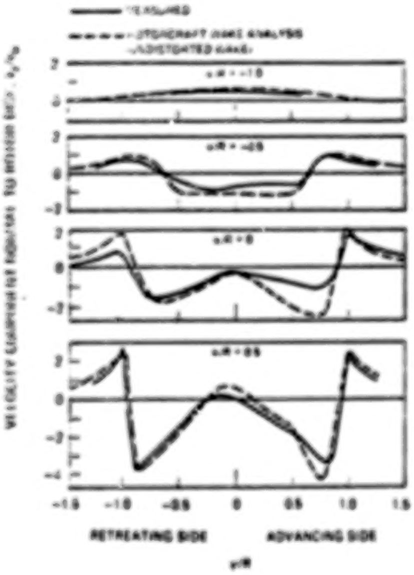
AIRLOADS AND WAKES - Continued

Example of rotor airloads calculations (Scully, 1975):



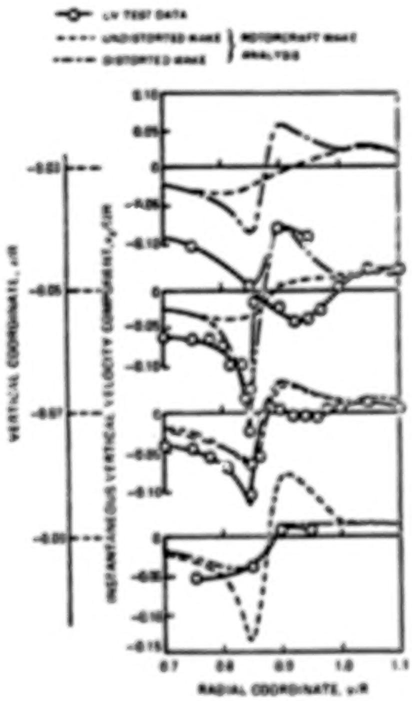
Four-blade rotor; μ = 0.18.

Example of rotor wake velocity calculations (Landgrebe and Egolf, 1976b):



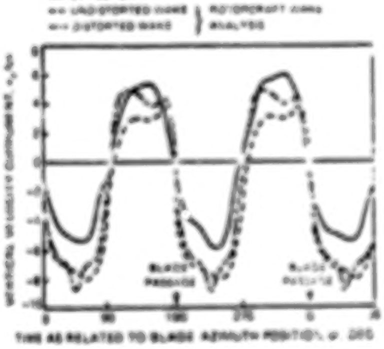
μ = 0.14; z/R = -0.07

Time-averaged induced velocity component beneath rotor.



V/ΩR = 0.18

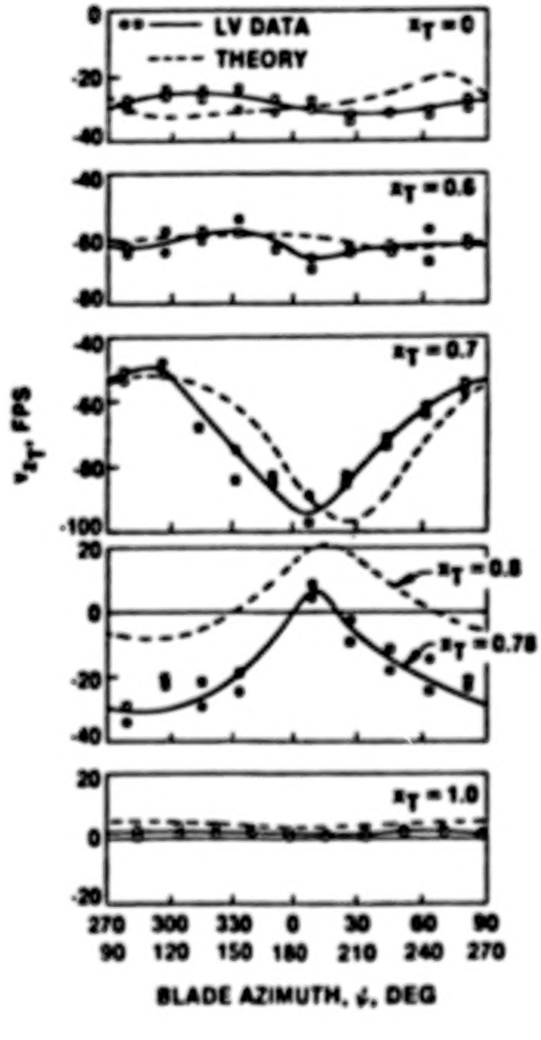
Radial distribution of instantaneous vertical velocity component beneath advancing blade of rotor.



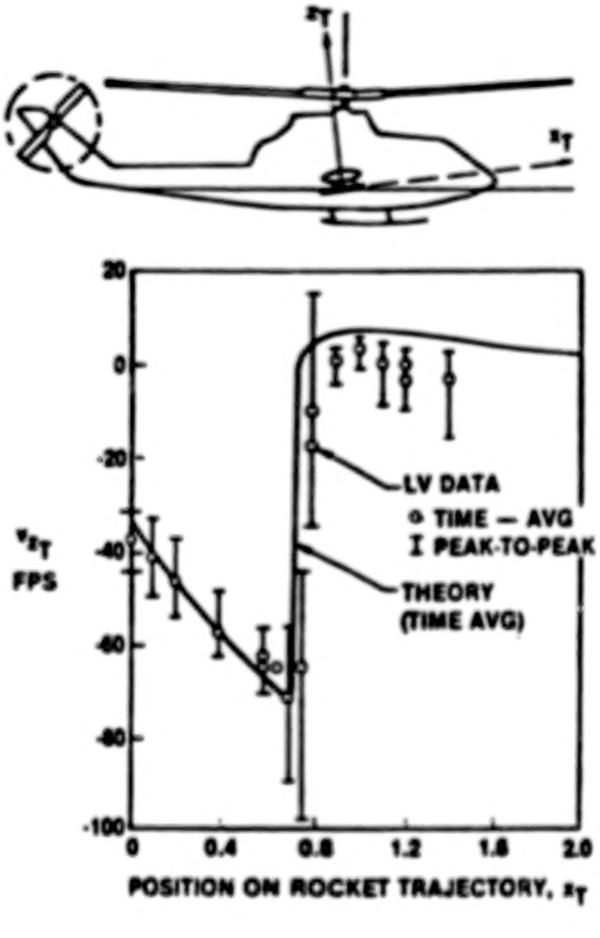
μ = 0.15

Time histories of vertical velocity components at fixed point near rotor.

Example of rotor wake velocity calculations (Landgrebe et al., 1981):

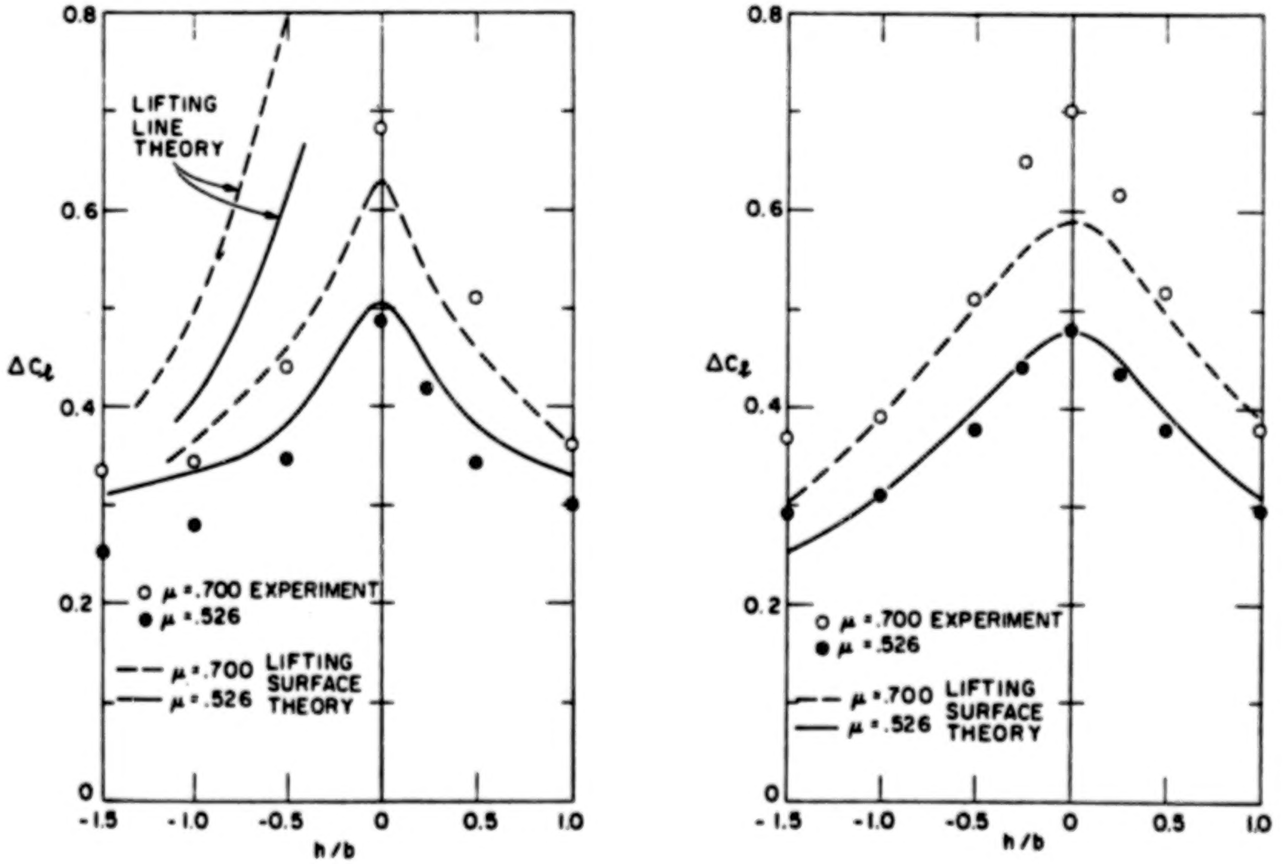


Time variant flow velocities for hover.



Flow velocities for hover.

Example of calculation of vortex induced blade loads (Johnson, 1971):

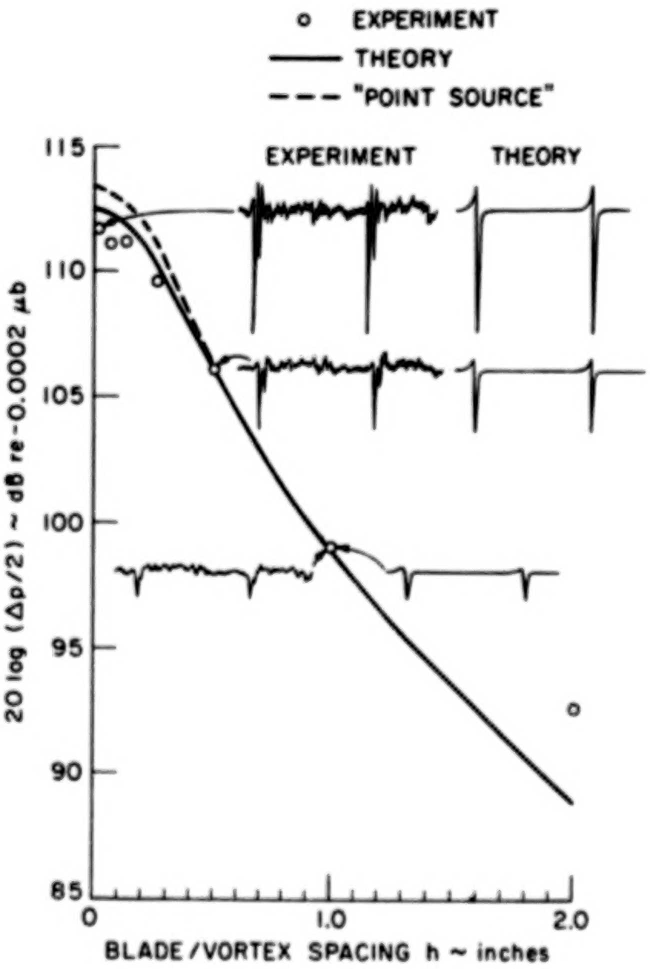


$y_G = -0.75$; $r = 0.95$

$y_G = 0.50$; $r = 0.95$

Peak-to-peak lift coefficient for vortex/blade interaction.

Example of calculations of blade/vortex interaction noise (Widnall, 1971):



Effect of blade/vortex spacing on transient signal.
Rotor rpm = 2000.

DYNAMIC STALL

The following figures show examples of attempts to calculate dynamic stall of airfoils.

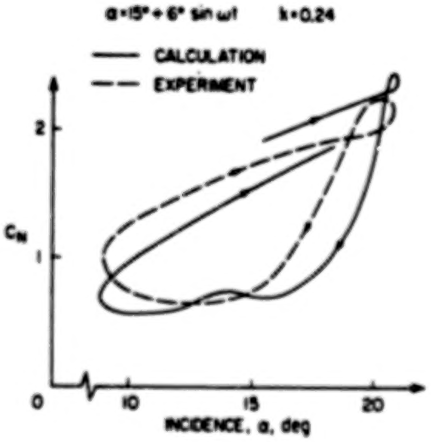
The models used include unsteady potential theory (Ham; Baudu); potential flow, including model of the separated wake, with a quasistatic boundary-layer solution and a time lag for dynamic stall effects (Rao); unsteady, incompressible, laminar Navier-Strokes calculations (Mehta); unsteady, compressible, Navier-Stokes calculations with turbulence models (Tassa and Sankar). The potential flow analyses can model the vortex shedding from the leading edge that characterizes dynamic stall and can predict the high transient loads; but they require a specification of when the dynamic stall occurs. The Navier-Stokes solutions so far have not produced quantitatively good results but have exhibited the qualitative characteristics of dynamic stall. The discrepancies are attributed to the turbulence model, as usual. Absence of correlation with experiment reflects the level of accuracy achieved so far.

Stall on the rotor is occasionally identified as a major source of noise. It is likely in addition that viscous effects are important in all aerodynamic phenomena producing rotor noise. For example, blade/vortex interaction may involve stall-like phenomena on the blade due to the high pressure gradients induced by the vortex or the blade may induce changes to the viscous core of the vortex.

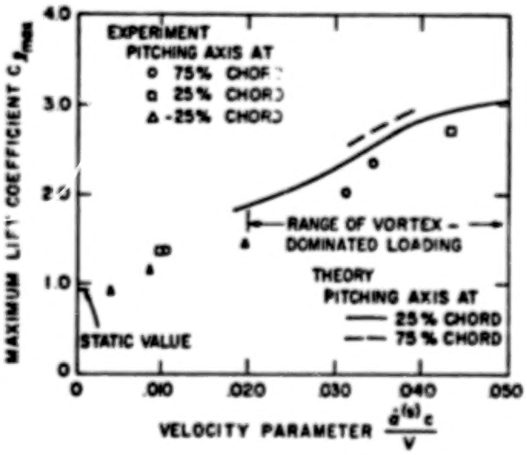
- EXAMPLES OF CAPABILITY TO PREDICT ROTOR BLADE STALL AERODYNAMICS
    - ---- MODELS RANGE FROM UNSTEADY POTENTIAL THEORY TO UNSTEADY, COMPRESSIBLE NAVIER-STOKES WITH TURBULENCE MODEL
    - ---- POTENTIAL FLOW ANALYSES CAN MODEL VORTEX SHEDDING EFFECTS BUT REQUIRE SPECIFICATION OF WHEN DYNAMIC STALL OCCURS
    - ---- NAVIER-STOKES SOLUTIONS HAVE SO FAR PRODUCED RESULTS THAT ARE QUALITATIVELY CORRECT
- IMPLICATIONS FOR NOISE CALCULATIONS
    - ---- STALL IS OCCASIONALLY IDENTIFIED AS AN IMPORTANT NOISE MECHNISM; VISCOUS EFFECTS FACTOR IN OTHER AERODYNAMIC PHENOMENA TOO

DYNAMIC STALL - Continued

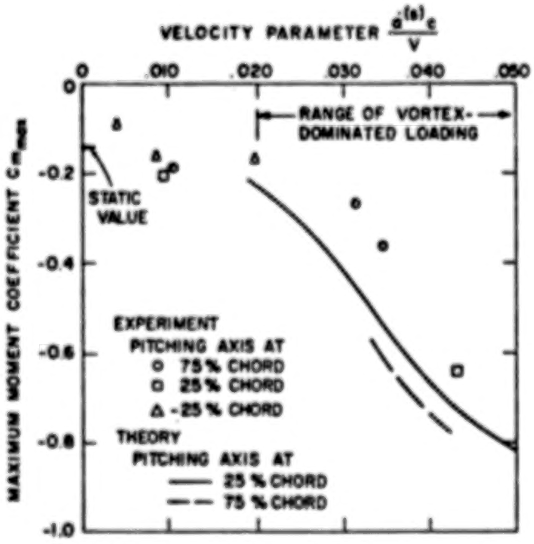
Examples of dynamic stall calculations using unsteady potential theories: top: maximum loads due to ramp increases in angle of attack (Ham and Garelick, 1968); bottom: lift due to oscillation in angle of attack (McCroskey, 1978):



Maximum lift coefficient
vs velocity parameter.



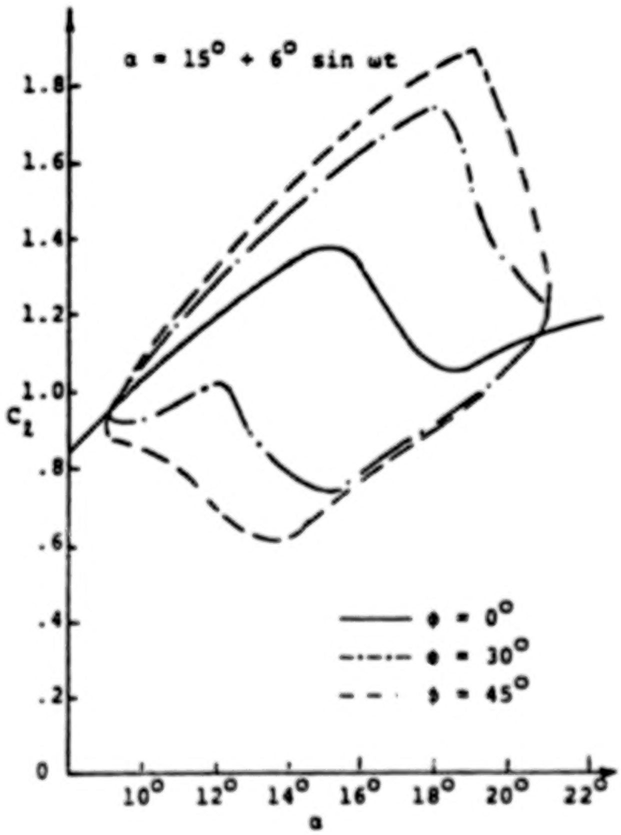
Maximum moment coefficient
vs velocity parameter.



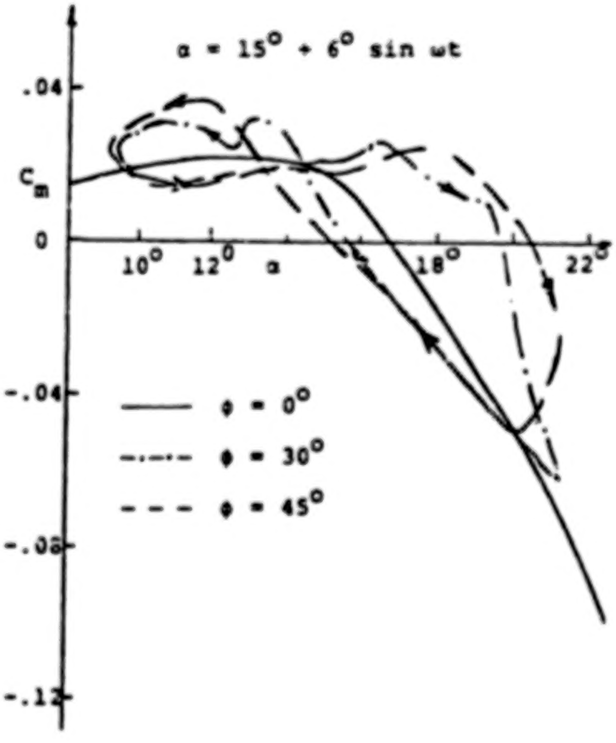
Comparison of calculations of Baudu et al., 1977
and measurements of Martin et al., 1974.

DYNAMIC STALL - Continued

Example of dynamic stall calculations (Rao et al., 1978); potential flow model includes separated wake, quasistatic boundary-layer analysis for separation points, and time lag for dynamic stall effects:



C_l vs α

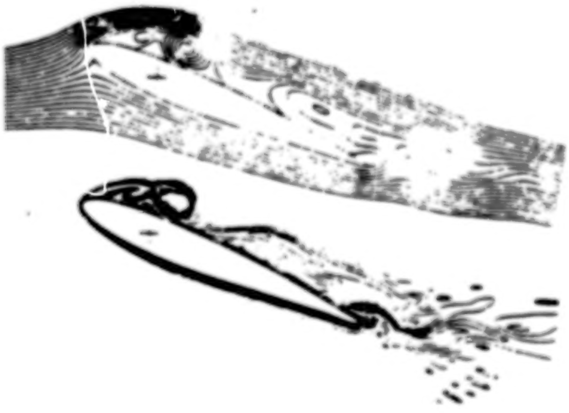


C_m vs α

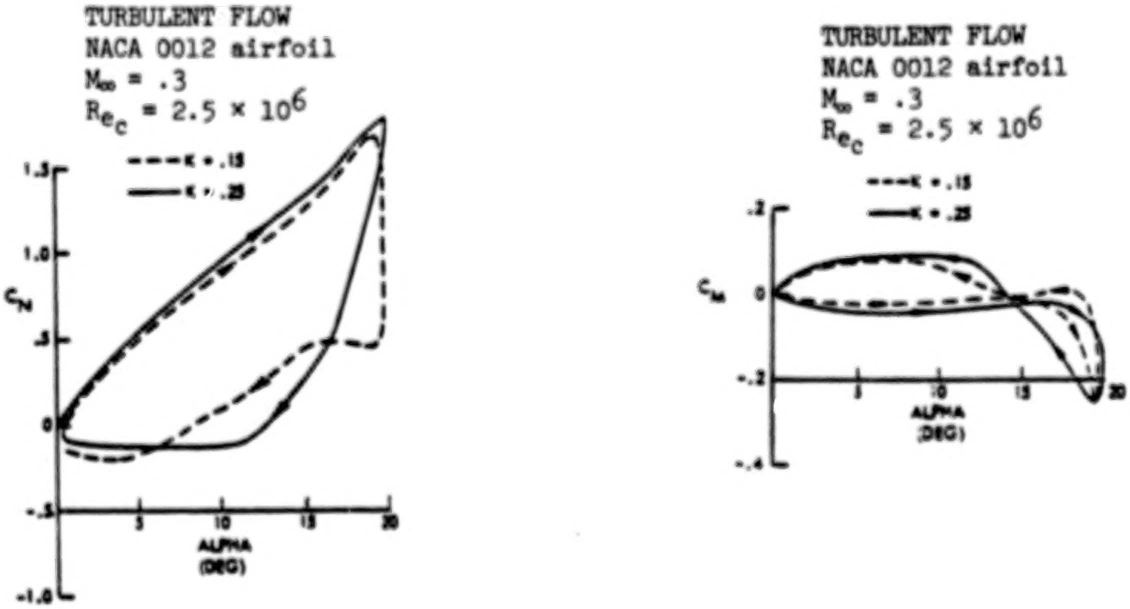
DYNAMIC STALL - Concluded

Examples of dynamic stall calculations using finite-difference analysis: top: unsteady, incompressible Navier-Stokes calculations for laminar flow at low Reynolds number (Mehta, 1978); bottom: unsteady, compressible Navier-Stokes calculations with turbulence model (Tassa and Sankar, 1981):

SECOND-ORDER NUMERICAL SCHEME



Streamlines and equivorticity lines.
$R = 5000$; $k = 0.5$; $\alpha = 20^\circ$.



Hysteresis loops of normal-force and moment coefficients.

COMPRESSIBLE FLOW

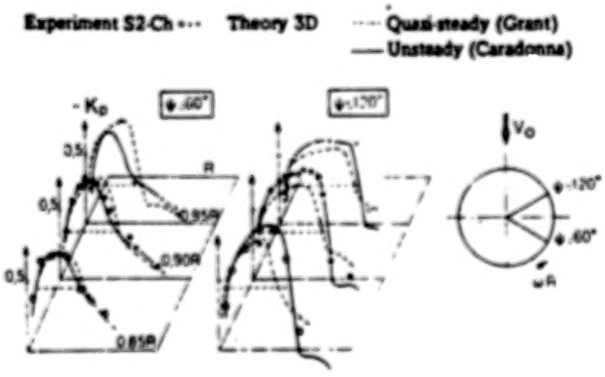
The following figures show examples of calculations of the loading on rotor blades in transonic flow, specifically at the advancing blade tip for high Mach numbers.

The inviscid, transonic potential flow equations are solved by finite-difference methods. Compared with a nominal model of small disturbance, unsteady, lifting, three-dimensional equations, the models being used involve further approximations: nonlifting (Caradonna; Chattot); two-dimensional (Caradonna); quasisteady (Grant); quasisteady but full potential (Arieli and Tauber). None of the analyses is coupled with the rotor wake or blade motion calculations; hence, the lifting solutions require a prescribed angle-of-attack variation. For the nonlifting case, the unsteady, three-dimensional, small disturbance solutions give good results compared with measurements. Calculations indicate that unsteady effects remain important for the lifting case. No measurements can be made to compare with the lifting calculations as long as the rotor wake is omitted from the analyses.

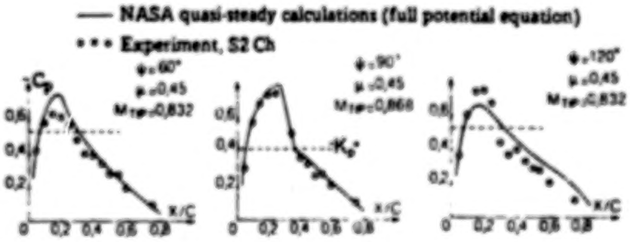
The analyses are being used with success to calculate rotor high-speed impulsive noise (basically a nonlifting phenomenon). A complete calculation of rotor noise requires a complete aerodynamics calculation; hence, ultimately the compressible flow solutions must be integrated with a rotor analysis including the wake, blade motion, and viscous effects.

- EXAMPLES OF CAPABILITY TO PREDICT ROTOR BLADE LOADING AT HIGH MACH NUMBER (ON THE ADVANCING BLADE TIP)
    - ---- FINITE DIFFERENCE SOLUTIONS OF INVISCID, TRANSONIC FLOW EQUATIONS
    - ---- ANALYSES SO FAR INVOLVE FURTHER APPROXIMATIONS: QUASISTEADY, OR NONLIFTING, OR SMALL PERTURBATION, OR TWO-DIMENSIONAL
    - ---- NONE OF ANALYSES COUPLED WITH ROTOR WAKE OR BLADE MOTION CALCULATIONS
    - ---- UNSTEADY, THREE-DIMENSIONAL CALCULATIONS GIVE GOOD RESULTS FOR NONLIFTING CASES
- IMPLICATIONS FOR IMPULSIVE NOISE CALCULATIONS
    - ---- ANALYSES ARE BEING USED WITH SUCCESS TO CALCULATE ROTOR HIGH SPEED IMPULSIVE NOISE
    - ---- USE IN COMPLETE NOISE CALCULATIONS WILL BE LIMITED UNTIL CAN MAKE COMPLETE AERODYNAMICS CALCULATIONS (COUPLING COMPRESSIBLE FLOW ANALYSES WITH WAKE AND MOTION SOLUTIONS)

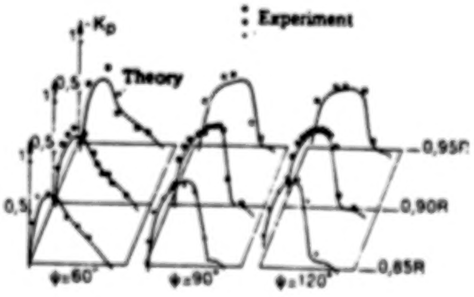
Examples of calculation of transonic loads on nonlifting rotor blade in forward flight (figures from Philippe and Chattot, 1980): top left: unsteady calculations (Caradonna and Isom, 1976); and quasisteady calculations (Grant, 1979); top right: quasisteady, full potential calculations (Arieli and Tauber, 1979); bottom left and right: unsteady calculations (Chattot, 1980):



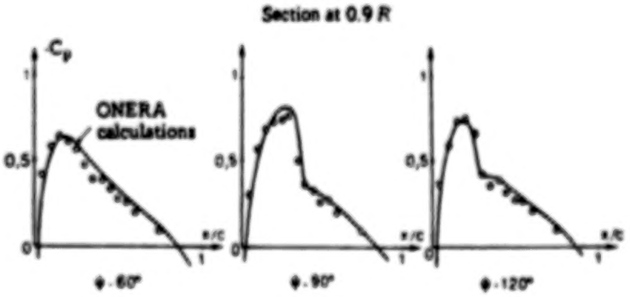
Nonlifting rotor; V_o = 110 m/s; wR = 200 m/s.



NASA calculations vs. experiment in S2 Ch wind tunnel.



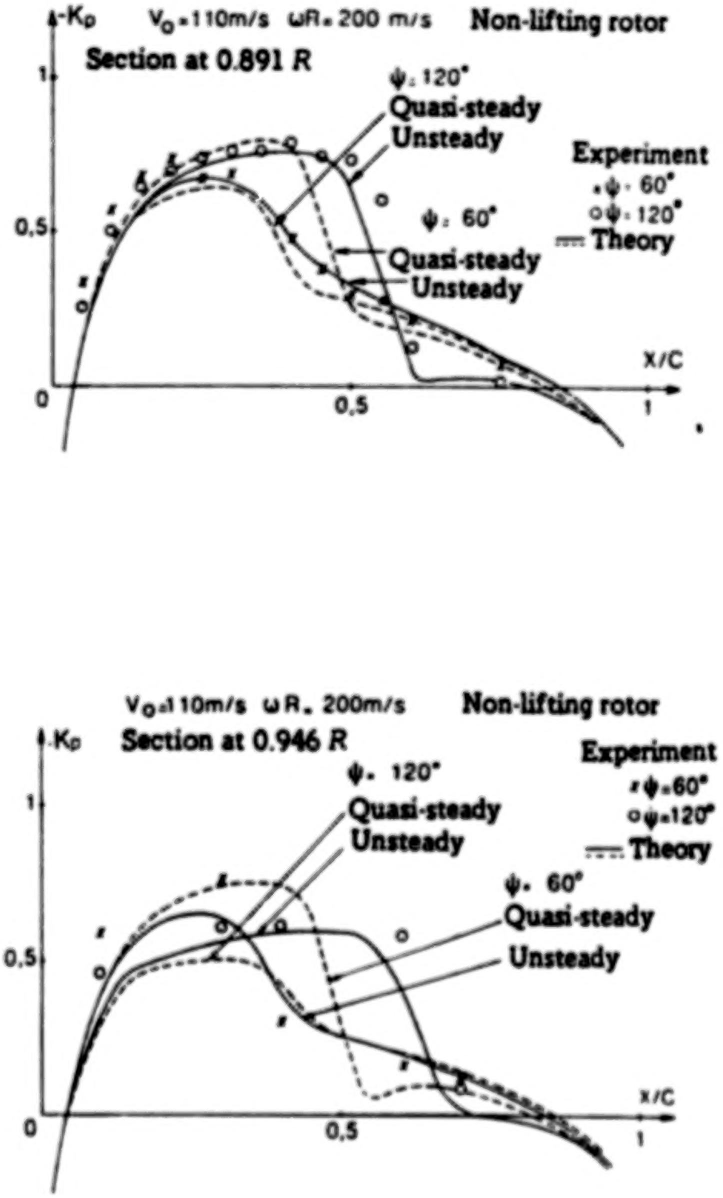
μ = 0.45; V_o = 90 m/s; wR = 200 m/s.



μ = 0.55; V_o = 110 m/s; wR = 200 m/s.

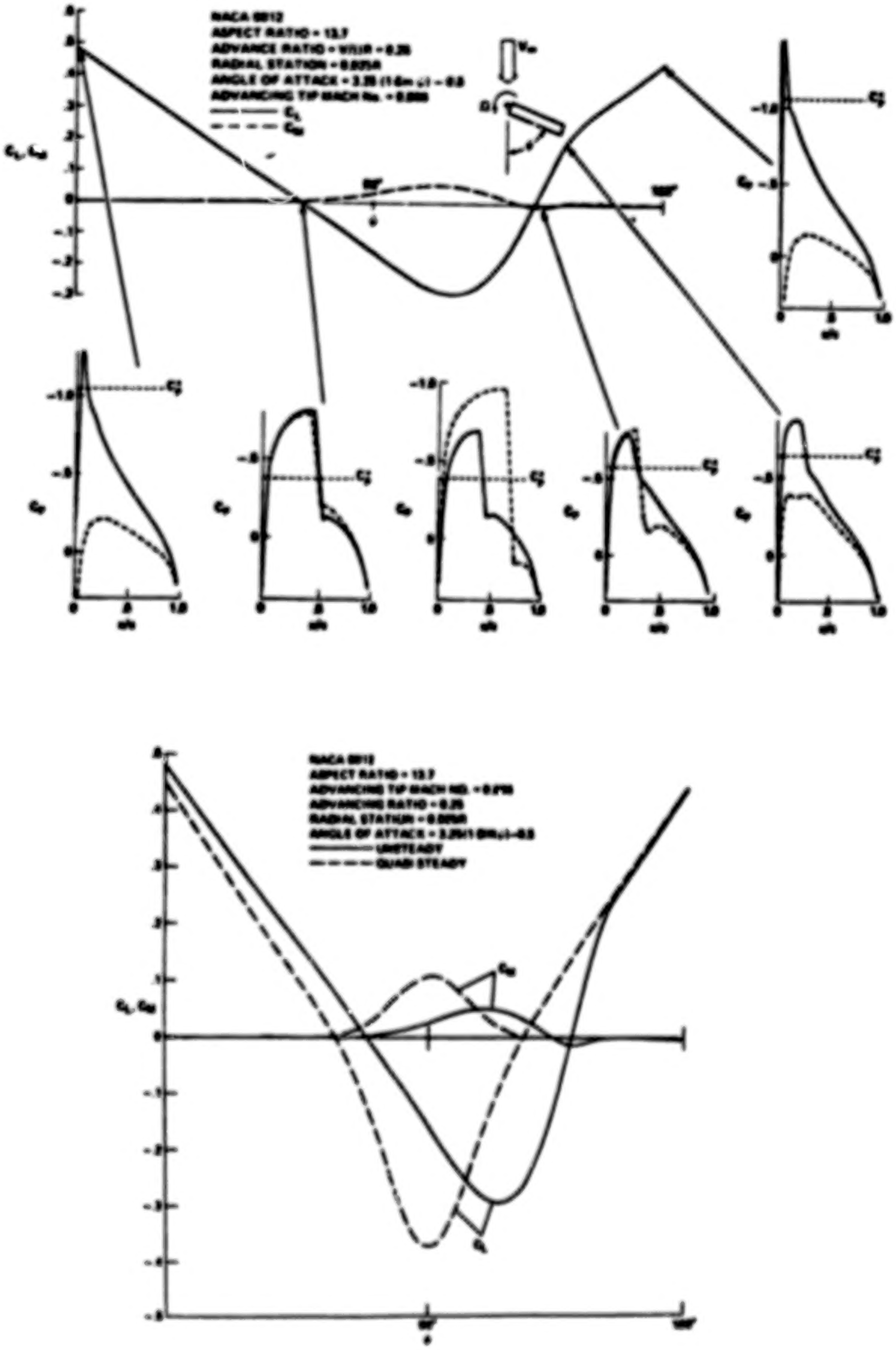
Pressure distributions on straight blade tips.

Example of calculation of transonic loads on nonlifting rotor blade, showing the influence of unsteady terms in the equations (Philippe and Chattot, 1980):



Unsteady and quasi-steady calculations vs. experiment.

Example of calculation of transonic loads on a two-dimensional lifting rotor blade with a prescribed angle-of-attack variation, with a comparison of unsteady and quasisteady calculations (Caradonna and Philippe, 1978):



Load variation on helicopter rotor.

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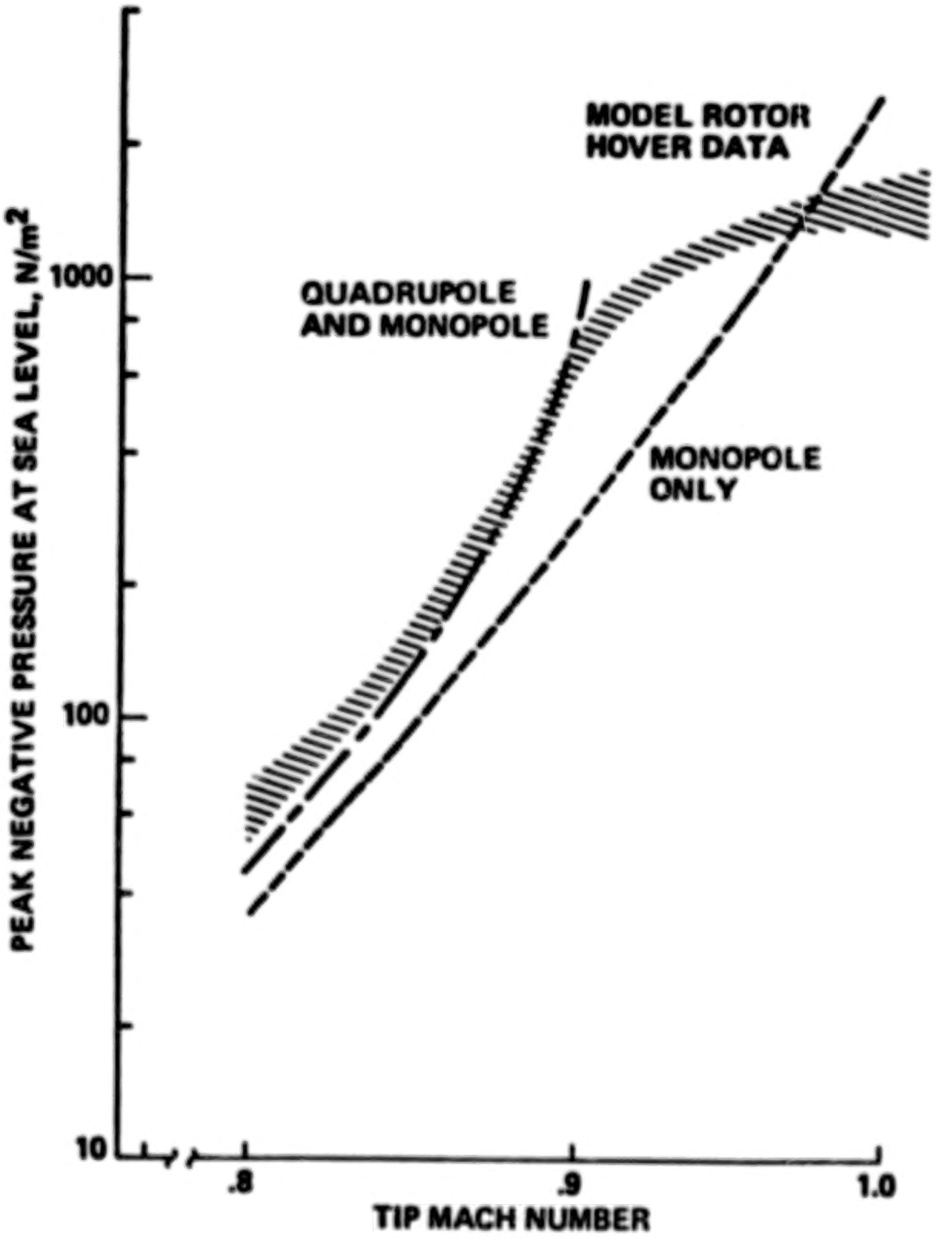
Example of high-speed rotor impulsive noise calculations, based on transonic flow calculations of Caradonna (Schmitz and Yu, 1980):



Comparison of theory and experiment in hover. In-plane peak acoustic pressure; $r/D = 1.5$.

RANDOM LOADS

The following figures show examples of calculations of rotor broadband noise. Two sources are considered: trailing-edge noise due to boundary-layer turbulence, and noise due to incident turbulence.

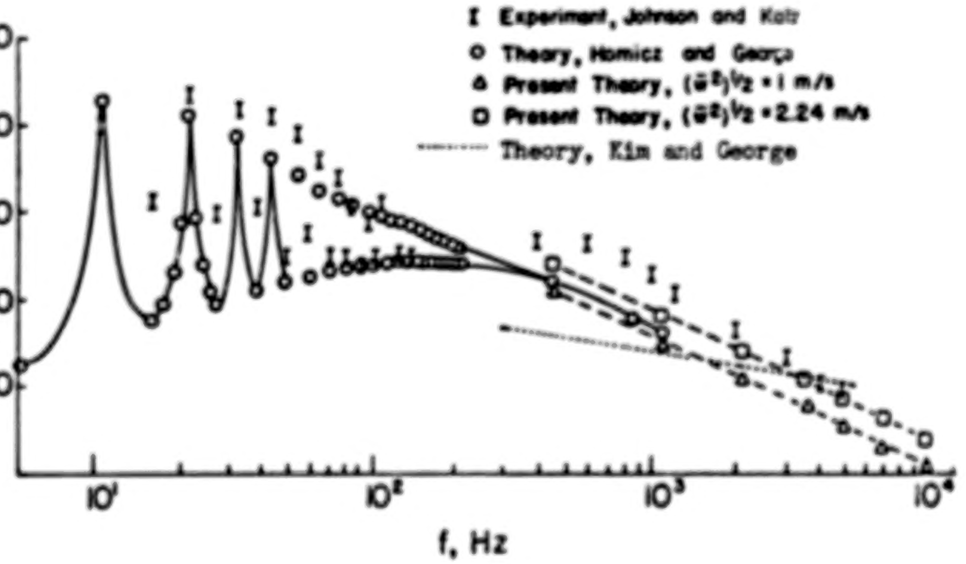
Extremely simple aerodynamic theories are being used in the current calculations of broadband noise. Noise due to boundary-layer turbulence is calculated using an empirical spectrum for the surface pressure, from measurements on a flat plate or airfoil. Noise due to incident turbulence is calculated using typically a Dryden gust spectrum, and a high frequency approximation for the compressible Sears function (from linear two-dimensional aerodynamic analysis) to calculate the loading. There are some measurements of the surface pressure spectra on an airfoil (nonrotating, two-dimensional). There are no measurements available of the random loading on a rotor blade.

The prediction of broadband noise may be expected to remain fair at best until more rigorous aerodynamic analyses, proven by correlation with rotor measurements, are available for the random loads.

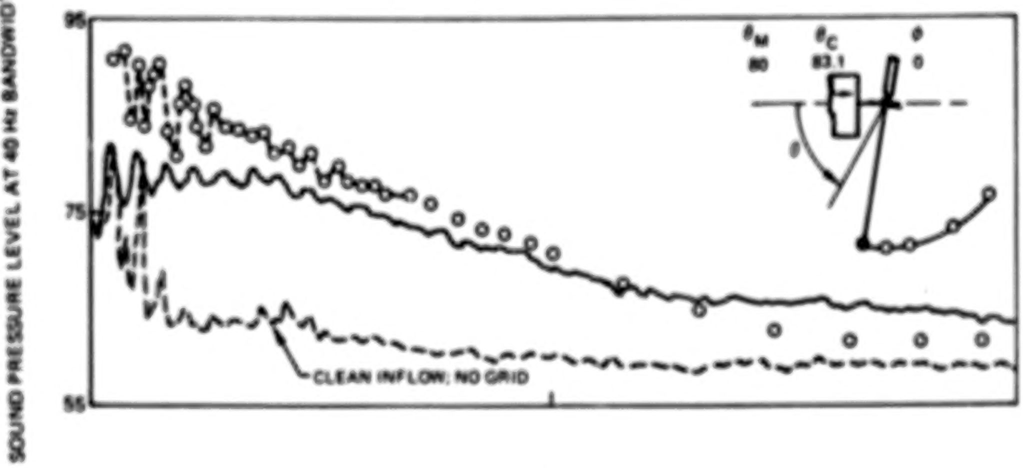
- EXAMPLES OF CALCULATIONS OF ROTOR BROADBAND NOISE AND RANDOM AERODYNAMIC LOADING
  - ---- TRAILING EDGE NOISE DUE TO BOUNDARY LAYER TURBULENCE; BASED ON EMPIRICAL SPECTRUM FOR SURFACE PRESSURE
  - ---- NOISE DUE TO INCIDENT TURBULENCE; BASED ON LINEAR AERODYNAMIC THEORY (EXTENDED SEARS FUNCTION) AND EMPIRICAL GUST SPECTRUM
  - ---- NO COMPARISONS WITH MEASURED ROTOR LOADING AVAILABLE
- IMPLICATIONS FOR BROADBAND NOISE CALCULATIONS
  - ---- FAIR PREDICTION OF NOISE HAS BEEN ACHIEVED; BUT THE AERODYNAMIC MODELS BEING USED ARE EXTREMELY SIMPLE

RANDOM LOADS - Continued

Example of calculations of rotor broadband noise: top: incident turbulence noise predictions (Homicz and George, 1974), high-frequency approximation for incident turbulence noise (George and Kim, 1977), and trailing-edge noise predictions (Kim and George, 1980); bottom: incident turbulence noise predictions (Paterson and Amiet, 1979):



Comparison of helicopter noise spectrum measurements of Johnson and Katz, 1972, with theoretical calculations using Dryden spectrum with undistorted flow and distorted flow turbulent intensities.



Rotor turbulence ingestion noise in simulated forward flight.
High turbulence level.

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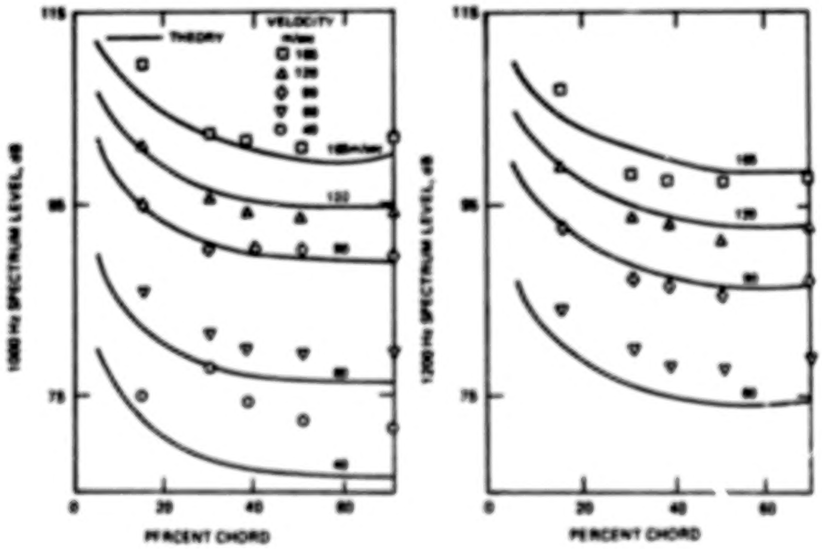
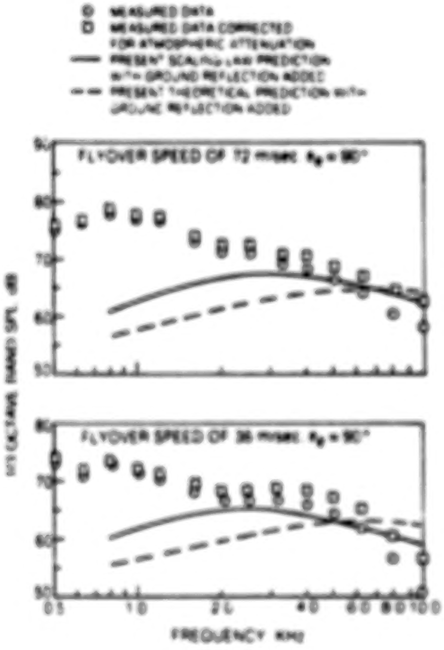
RANDOM LOADS - Concluded

Example of calculations of rotor broadband loads and noise: top: surface pressure spectra due to incident turbulence (Paterson and Amiet, 1977); bottom: prediction of rotor trailing-edge noise (Schlinker and Amiet, 1981):



Measured vs predicted surface pressure distributions.



Measured vs predicted trailing-edge noise
for helicopter flyover.

DYNAMICS

It must be remembered that almost all phenomena of helicopters are aeroelastic problems, involving the coupling of aerodynamic, inertial, and structural loads. The aerodynamic analyses can not be considered in isolation, since ultimately they require an angle of attack, which can only be obtained by a fully coupled solution for the blade motion and loading. The blade motion required may be simply the rigid flap motion, or it may include elastic motion (such as torsion deflection). It has often been attempted to obtain the blade motion for an aerodynamic analysis from another source (calculation or measurement); this approach has not been successful because the coupling between the loading and motion is not in just one direction. A fully coupled, fully consistent solution can only be obtained by a complete aeroelastic analysis.

It is useful to consider the research program in hingeless rotor dynamics as a possible analogy for the expanded rotor noise research program. Around 1970, research requirements were focussed on the stability problems of hingeless rotors. Several groups of capable persons in government, industry, and universities have been engaged in this research. Very significant progress has been made on the problem as a direct result of the expanded research. There are also still fundamental questions to be answered and important problems to solve. Moreover, as knowledge was acquired about the rotor dynamics, the scope of the investigations increased. This analogy can provide a guide for the level of resources required for the noise research program.

- HELICOPTER PHENOMENA ARE AEROELASTIC PHENOMENA
  - ---- AERODYNAMIC ANALYSES REQUIRE ANGLE OF ATTACK, WHICH CAN ONLY BE OBTAINED BY FULLY COUPLED SOLUTION FOR BLADE MOTION AND LOADING
- ANALOGY WITH RESEARCH PROGRAM IN HINGELESS ROTOR DYNAMICS
  - ---- MORE THAN A DECADE OF RESEARCH, BY SEVERAL GROUPS OF PEOPLE, ON A WELL DEFINED PROBLEM
  - ---- SIGNIFICANT PROGRESS HAS BEEN MADE; THERE ARE STILL FUNDAMENTAL QUESTIONS TO BE ANSWERED AND IMPORTANT PROBLEMS TO SOLVE
  - ---- AS KNOWLEDGE HAS BEEN ACCUMULATED, THE SCOPE OF THE INVESTIGATIONS HAS BEEN EXPANDED

AERODYNAMIC PREDICTIONS

There have been many major advances in the aerodynamic prediction capability for rotorcraft over the last decade. Advanced computational techniques are being applied to several parts of the rotor aerodynamic problem. There have been relatively few attempts to apply these solutions in the context of a fully coupled rotor analysis. These advances in the aerodynamic technology have been directly responsible for several improvements in the noise prediction capability.

The work on airloads and wakes, dynamic stall, compressible flow, and random loads is not complete. The development of this aerodynamic prediction methodology must continue. The direction the development must take is generally clear from the review of the present capability.

More attention must be given to the noise due to the interaction of the aircraft components, such as the main rotor and tail rotor. While much attention is presently being given to interactional aerodynamics, relatively little work has been done on the unsteady aerodynamics involved in the phenomena. The unsteady aerodynamic forces must be calculated not only for the noise prediction but also for vibration and oscillatory structural loads.

The aerodynamic methodology must be integrated into a complete theory for prediction of helicopter behavior. While the complete theory in certain cases must wait for further development of the aerodynamic (or dynamic) technology, the development of the techniques for the integration need not wait. A complete solution for the helicopter behavior is also of value even when it is based on methodology that is still being developed.

- CURRENT CAPABILITY
  - ---- THERE HAVE BEEN MANY MAJOR ADVANCES IN THE AERODYNAMIC PREDICTION CAPABILITY OVER THE LAST DECADE
  - ---- ADVANCED COMPUTATIONAL TECHNIQUES ARE BEING APPLIED TO A NUMBER OF ASPECTS OF ROTOR AERODYNAMICS
- FUTURE PROGRESS
  - ---- THE DEVELOPMENT OF PREDICTION METHODOLOGY FOR AIRLOADS AND WAKES, DYNAMIC STALL, COMPRESSIBLE FLOW, AND RANDOM LOADS IS NOT COMPLETE
  - ---- MORE ATTENTION MUST BE GIVEN TO THE NOISE DUE TO INTERACTION OF THE AIRCRAFT COMPONENTS (SUCH AS MAIN ROTOR AND TAIL ROTOR) AND THE UNSTEADY AERODYNAMICS REQUIRED TO PREDICT THE NOISE
  - ---- THE AERODYNAMIC METHODOLOGY MUST BE INTEGRATED INTO A COMPLETE THEORY FOR PREDICTION OF HELICOPTER BEHAVIOR

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AERODYNAMIC PREDICTIONS II

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The objective of the present work is the development of technology to aid in the design of advanced rotor blades. The goal is to improve both aerodynamic and acoustic performance. The main thrust of the effort is the development of advanced computer codes to calculate the subsonic and transonic flow field about rotor blades and to model such high noise producing phenomena as shock waves and blade/vortex interaction. In addition, the codes are verified by comparison with experimental data when available. Emphasis is also placed on making the codes available to industry and to provide some training for the industrial users.

OBJECTIVE

- DEVELOP TECHNOLOGY FOR ADVANCED ROTOR DESIGN

APPROACH

- CODE DEVELOPMENT
- EXPERIMENTAL VERIFICATION
- INDUSTRY INTERACTION

The aerodynamic research branch at Ames has extensive experience in transonic fixed-wing aerodynamics and has applied that expertise to rotary wings. A major achievement of this effort has been the development of the ROT22 computer code. The features and limitations of this code will be briefly described. The full three-dimensional transonic potential equation is solved in a blade-attached coordinate system. Lifting blades can be treated both in hover and forward flight and there are no limitations on the blade geometry. Using a mesh size of 120 × 16 × 24 (chordwise, vertically, and radially, respectively) results in a computation time of about 5 minutes on a CDC 7600 computer for the flow about the blade at one azimuthal position. The code also has provisions for including two straight vortices in the flow field at specified locations. Typically, one vortex is placed approximately normal to the leading edge to represent a tip vortex shed by a preceding blade to model the effect of inflow. The second vortex can be used to represent a high noise condition blade/vortex interaction, wherein the vortex is nearly parallel to the blade. The inclusion of the vortices increases the computation time by about 35 percent. A simple kinematic wake model is also included in the program.

The limitations of the code include the assumption of an inviscid flow field and the omission of the transient terms in the governing equations. The latter has the effect of eliminating both the "lag" and "memory" of the flow field; the blade has a fully developed flow at each azimuth and does not "remember" what the flow was at a previous azimuth angle. This quasi-steady assumption deteriorates with increasing advance ratio, but is exact in hover.

ROT22

FEATURES

- FULL POTENTIAL EQUATIONS (3-D)
- TRANSONIC
- LIFTING
- FORWARD FLIGHT AND HOVER
- REALISTIC GEOMETRIES
- FAST-5 min ON CDC 7600
- SIMPLIFIED BLADE-VORTEX INTERACTIONS
- PRESCRIBED NEAR WAKE

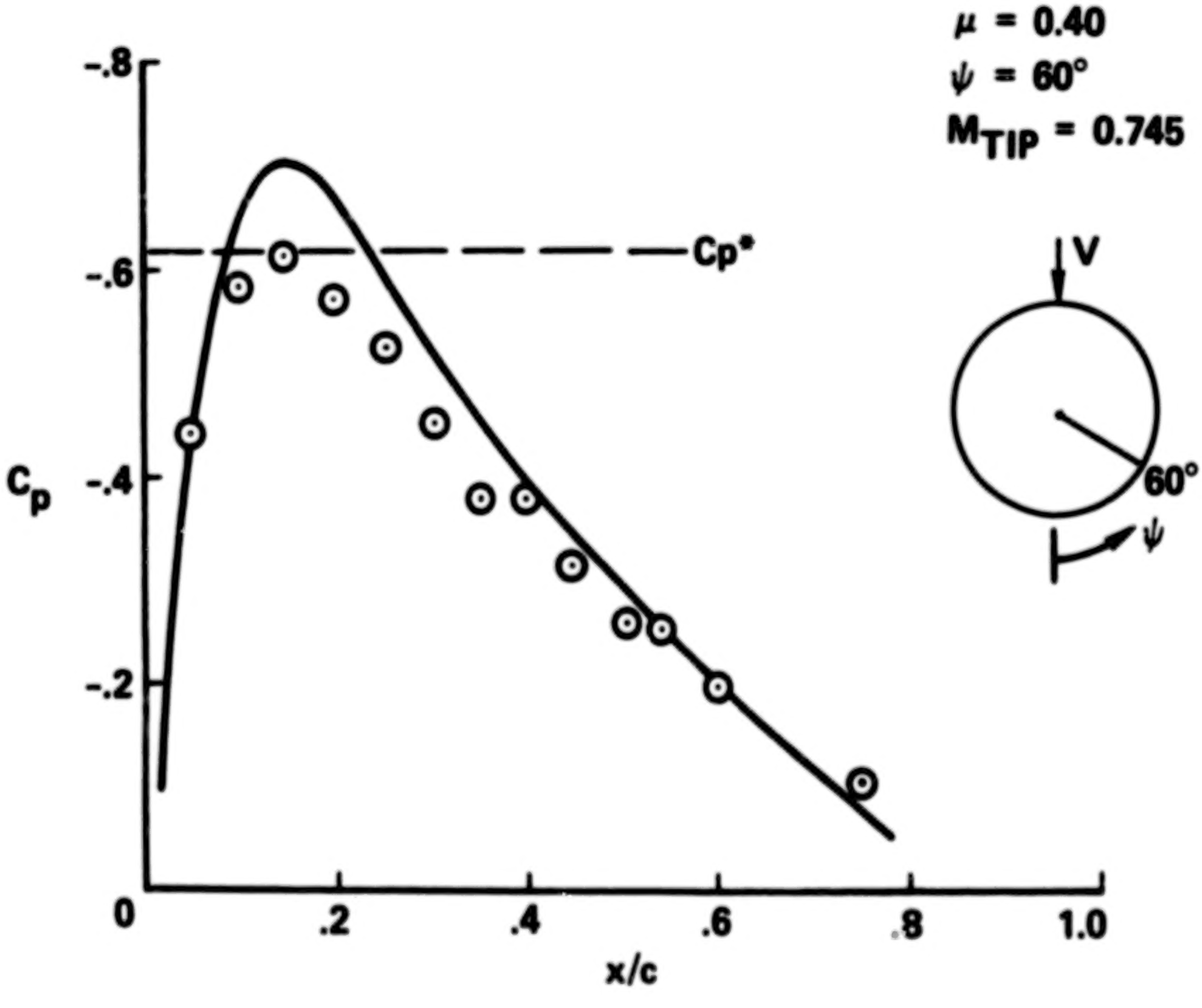
LIMITATIONS

- INVISCID
- QUASI-STEADY

To assess the effect of the quasi-steady assumption, the calculated and measured three-dimensional pressure distributions will be compared at one radial location on an alouette tail rotor tested at an advance ratio of 0.4. The rotor had symmetric airfoil sections and was not lifting. The chordwise pressures are compared at the best instrumented radial station of 0.892. At an azimuth angle of 60°, the quasi-steady calculation only moderately overpredicts the suction peak.

COMPARISON OF MEASURED AND CALCULATED PRESSURES

r/R = 0.892



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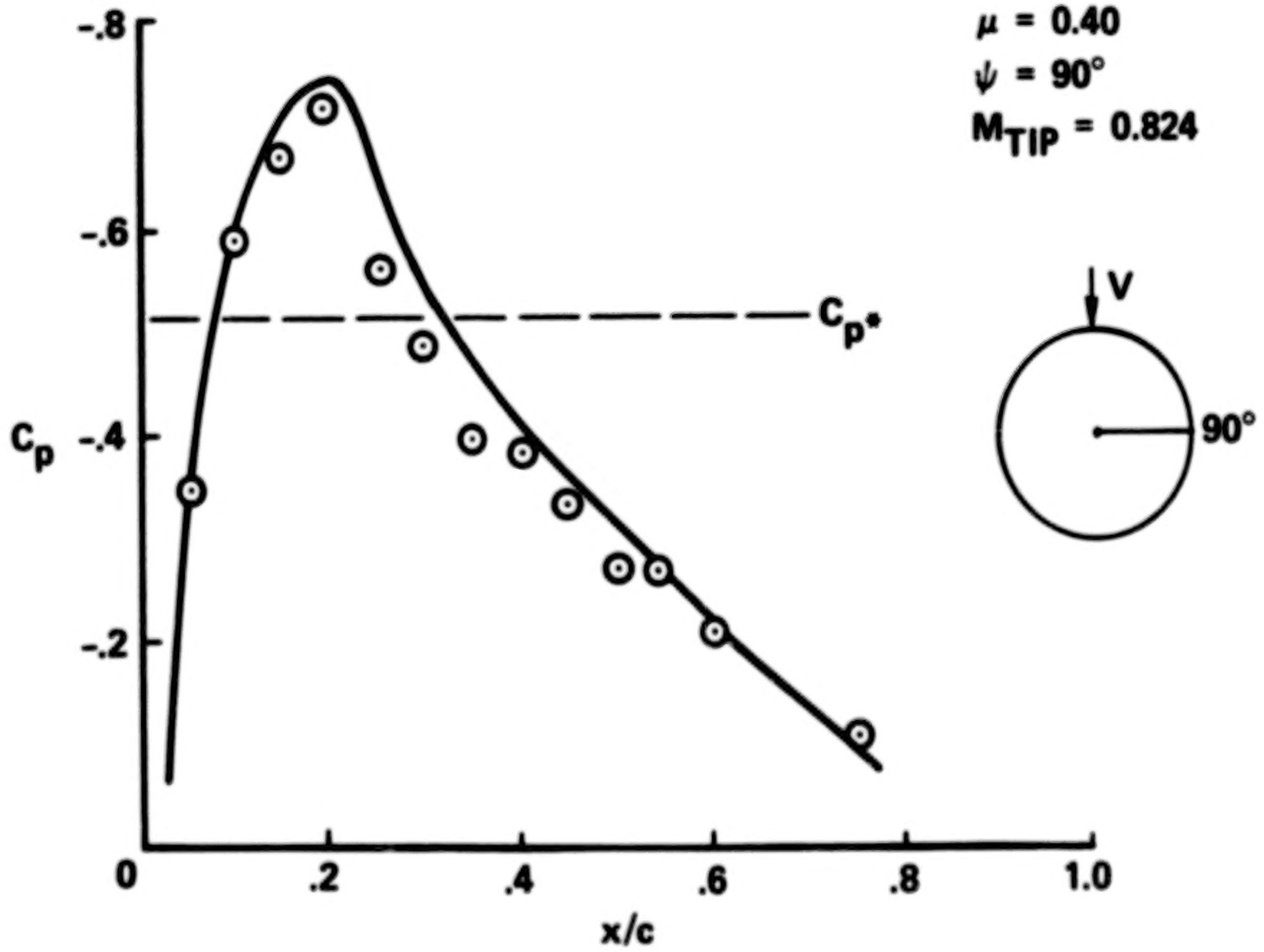
At 90° azimuth, the agreement between theory and experiment is very good.

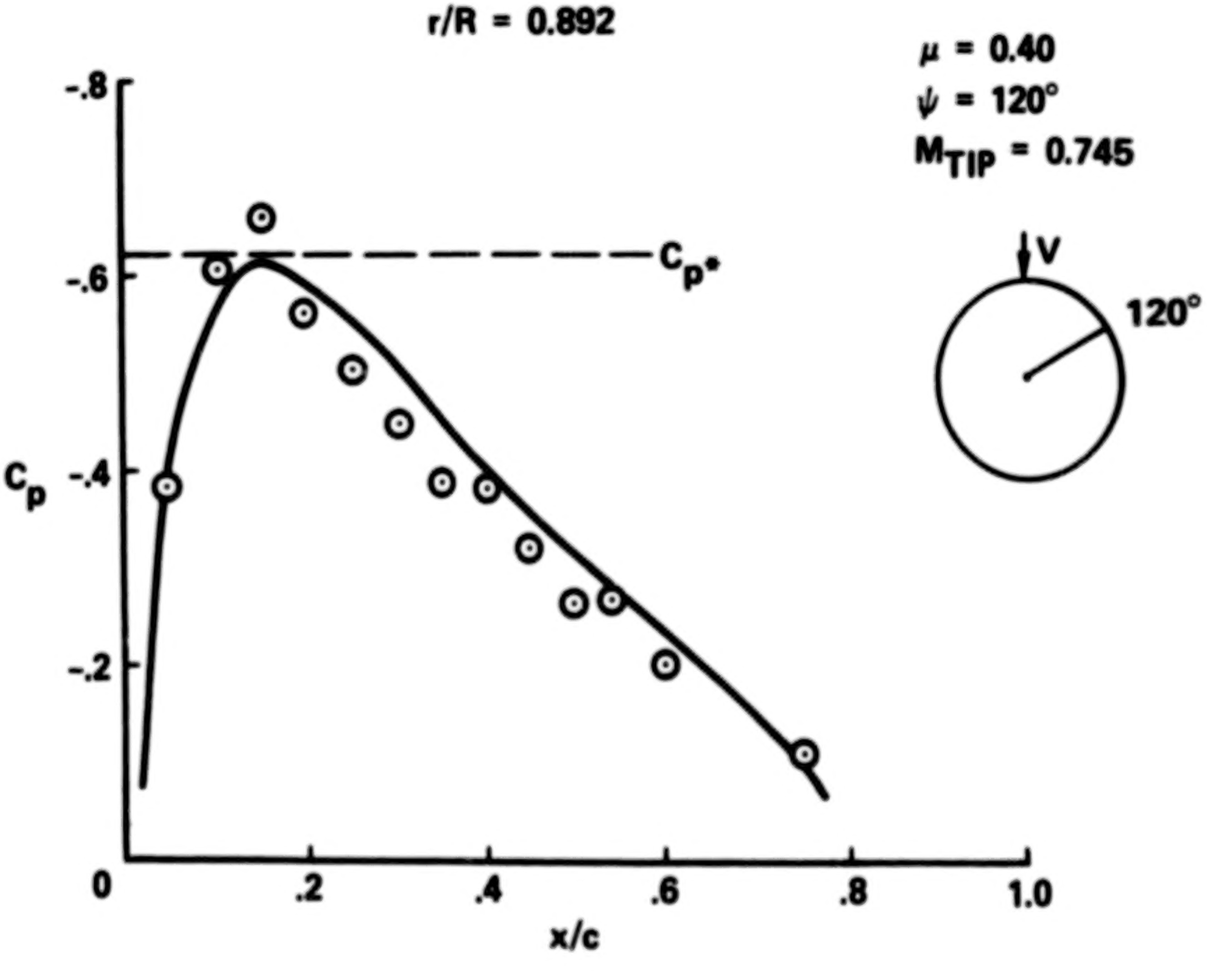
COMPARISON OF MEASURED AND CALCULATED PRESSURES

r/R = 0.892



In the second quadrant, at 120° the calculation underpredicts the suction peak somewhat. The differences between theory and experiment are attributed to the omission of the transient terms in the equations.

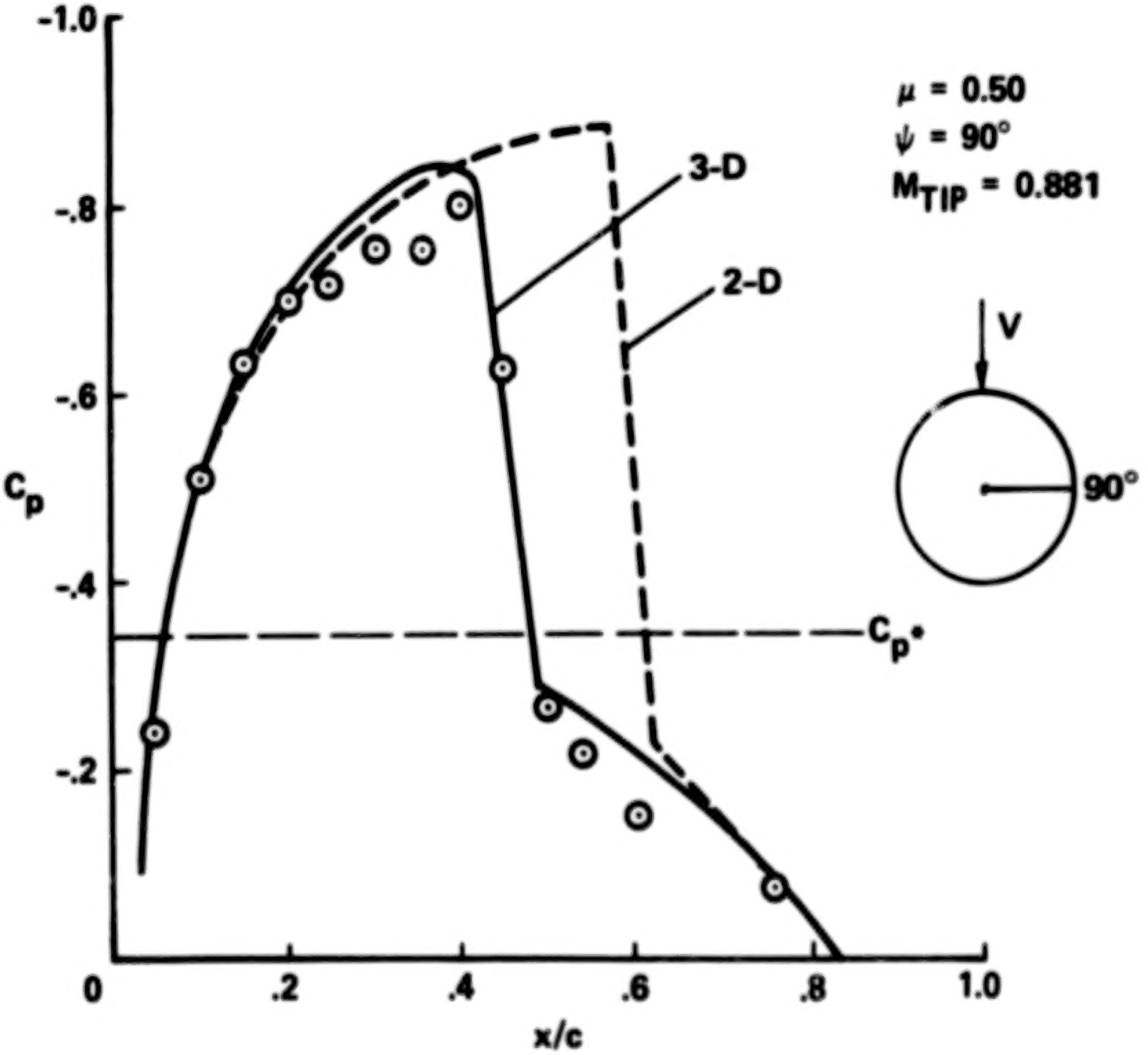
COMPARISON OF MEASURED AND CALCULATED PRESSURES



At 90° azimuth the calculated pressures even agree well with data from the same experiment at an advance ratio of 0.5. Both the shock strength and location are well predicted by the present three-dimensional calculations. In contrast, a two-dimensional calculation gives poor agreement at this radial station which is about 2/3 of a chord inboard of the tip.

COMPARISON OF 2-D AND 3-D CALCULATIONS WITH EXPERIMENT

r/R = 0.892

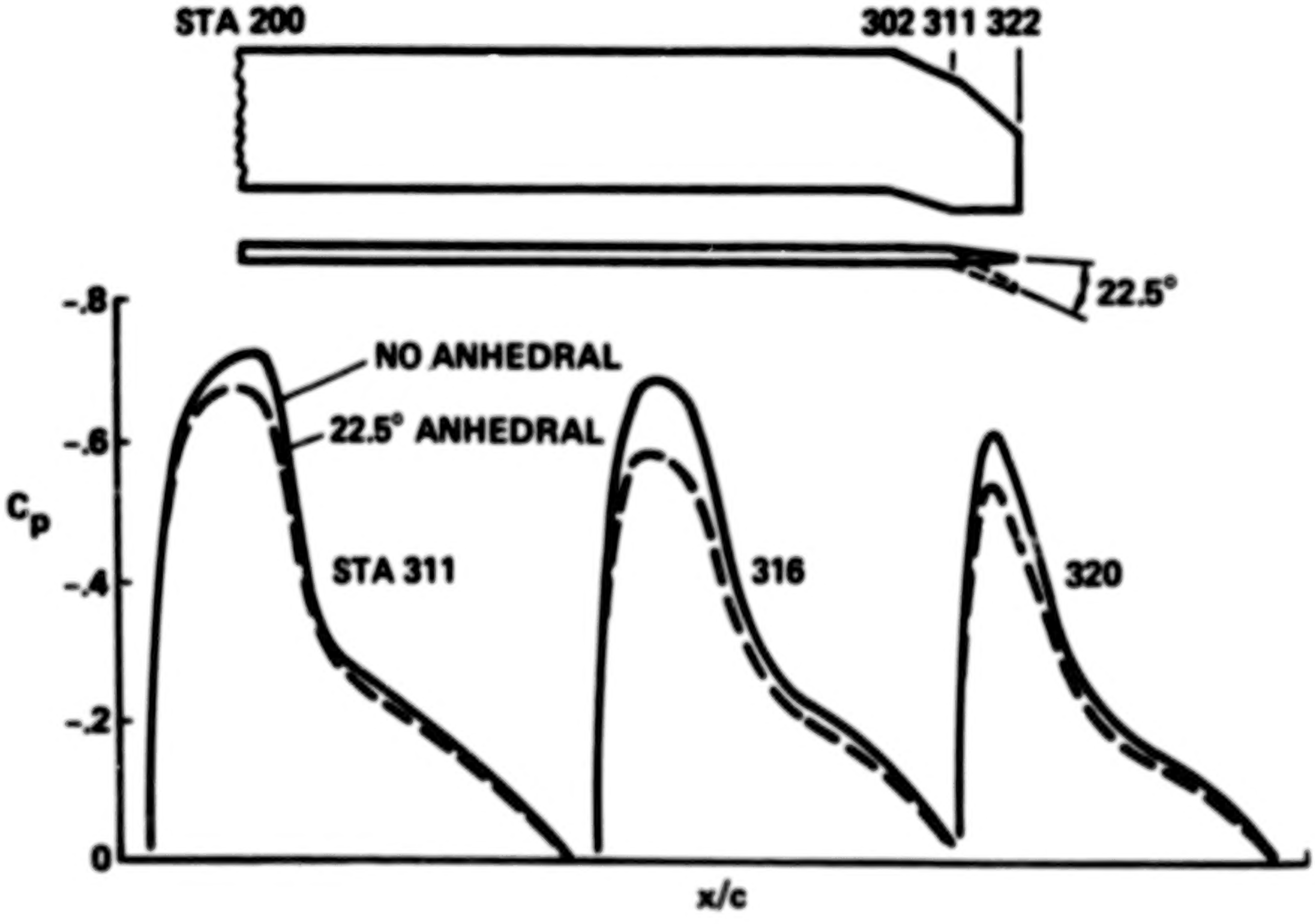


As another example of ROT22 code versatility, the code has been used to calculate the effect of introducing anhedral in the outer 3.5 percent of an actual lifting blade. The calculation was made for an advance ratio of 0.34 and azimuth angle of 90°. Although the flow field over the entire blade was computed, the code has provisions for increasing the grid density in critical regions, such as the blade tip, without increasing computation time. The effect on the upper surface pressure of introducing anhedral near the tip is clearly shown; lower surface pressures were omitted for clarity.

EFFECT OF TIP ANHEDRAL ON BLADE TOP SURFACE PRESSURE

SIKORSKY BLACKHAWK

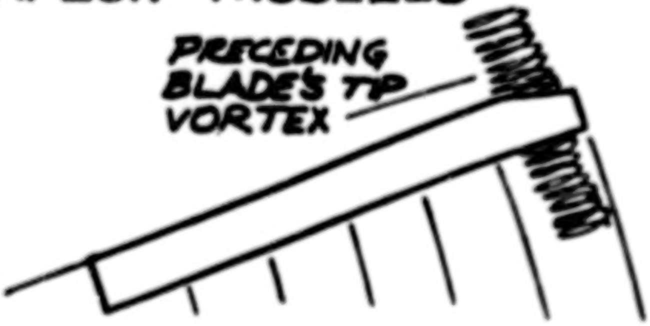
V = 145 knots, ψ = 90°



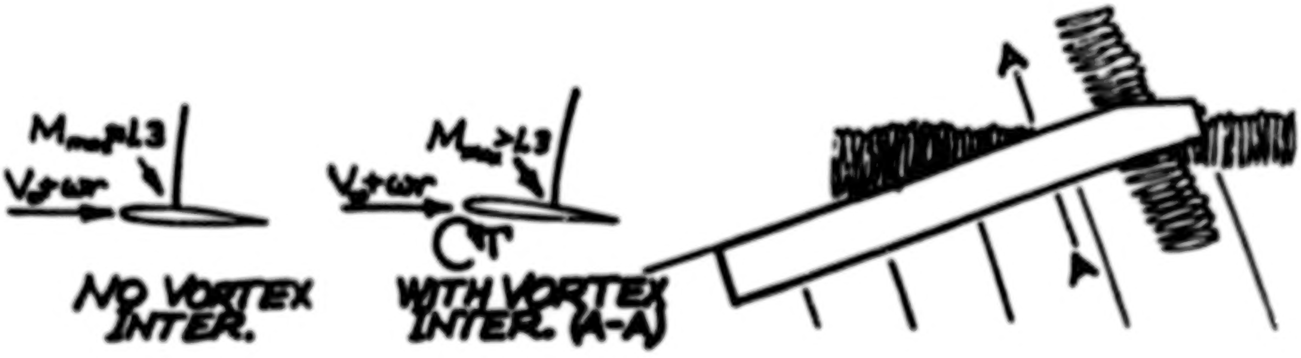
Work on a fully unsteady three-dimensional potential code is under way. While such a program would be more accurate than the quasi-steady code, computation time will be greater. As was previously mentioned, the capability to include two vortices has recently been added to the code to represent the influence of previously shed tip vortices. The effect on the flow of a vortex located roughly parallel to the leading edge is qualitatively illustrated at the bottom of the figure below. The quantitative effect is presented in the following figures.

ROT22 ENHANCEMENTS

- WORK ON FULLY UNSTEADY VERSION INITIATED
- SIMPLE VORTEX/INFLOW MODELED

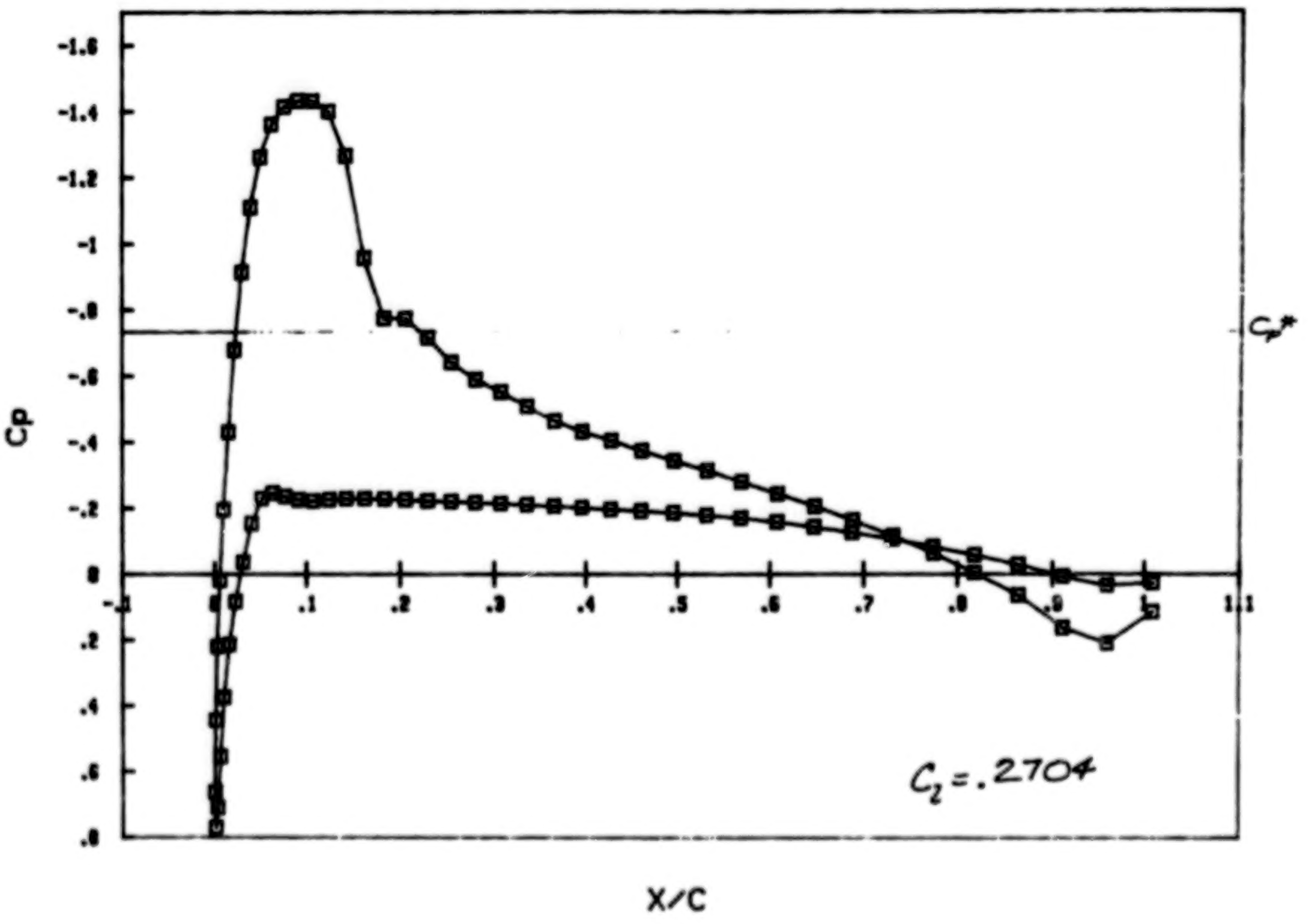

PRECEDING
BLADE'S TIP
VORTEX

- MULTIPLE VORTEX MODELING FINISHED – STUDY HIGH-SPEED NOISE MECHANISMS

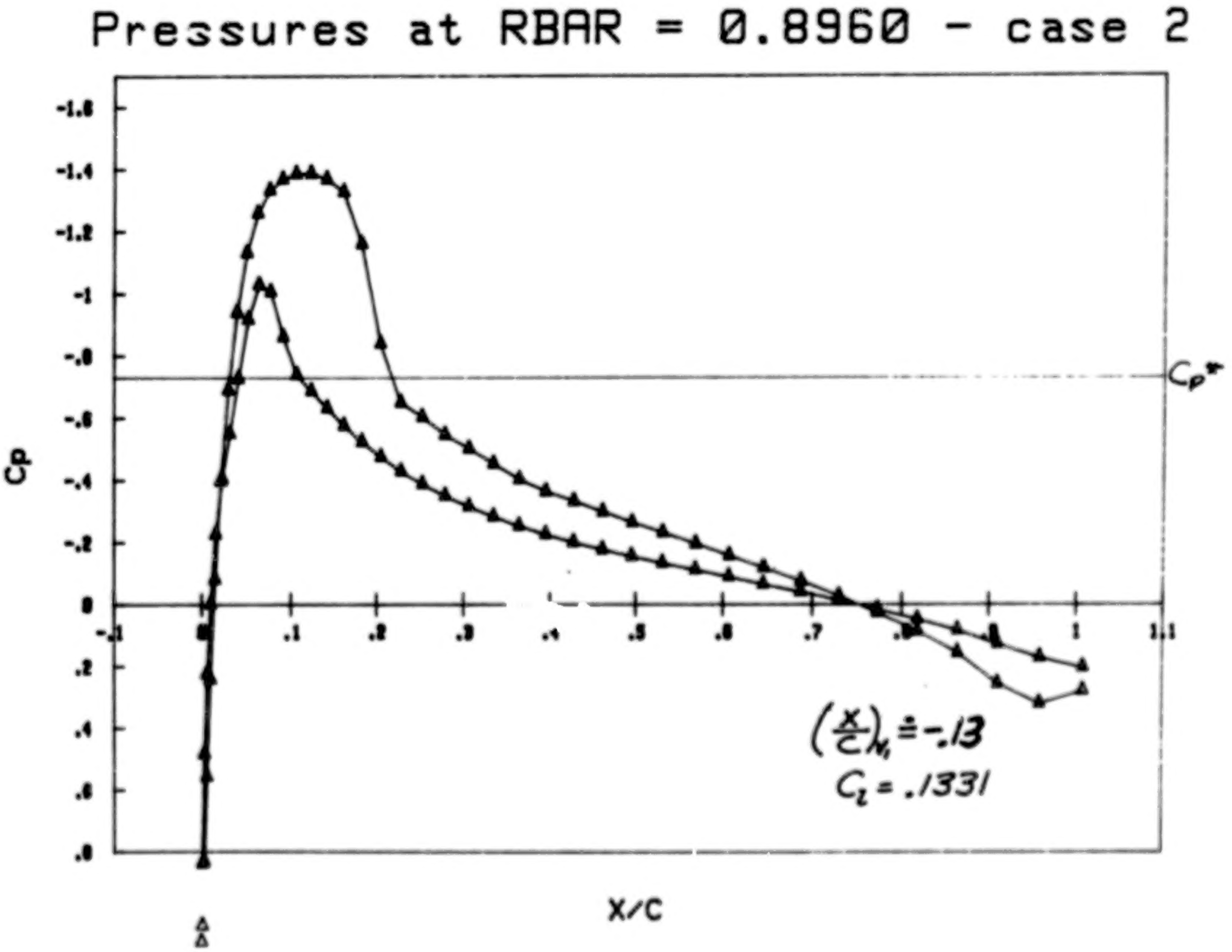

$M_{max} \approx 1.3$
$V_\infty + \omega r$
NO VORTEX INTER.
$M_{max} > 1.3$
$V_\infty + \omega r$
C_T
WITH VORTEX INTER. (A-A)
A
A

The flow field about an actual blade having two different airfoil sections, a swept and tapered tip, twist, and 2° of collective pitch was computed to study blade/vortex interaction. A low advance ratio of 0.15 was chosen so that one vortex would be roughly parallel to the blade at a blade azimuth of 60°. First, the flow field was calculated using only one vortex placed normal to the blade, one-half chord below at a radial station of 0.885, to simulate inflow. While all calculations were three-dimensional, pressures at only one radial position, 0.896, will be shown for simplicity. Note that the lift coefficient at this station is 0.27 and that a zone of supersonic flow exists on the upper surface, but that the flow on the lower surface is completely subsonic.

Pressures at RBAR = 0.8960 - case 1

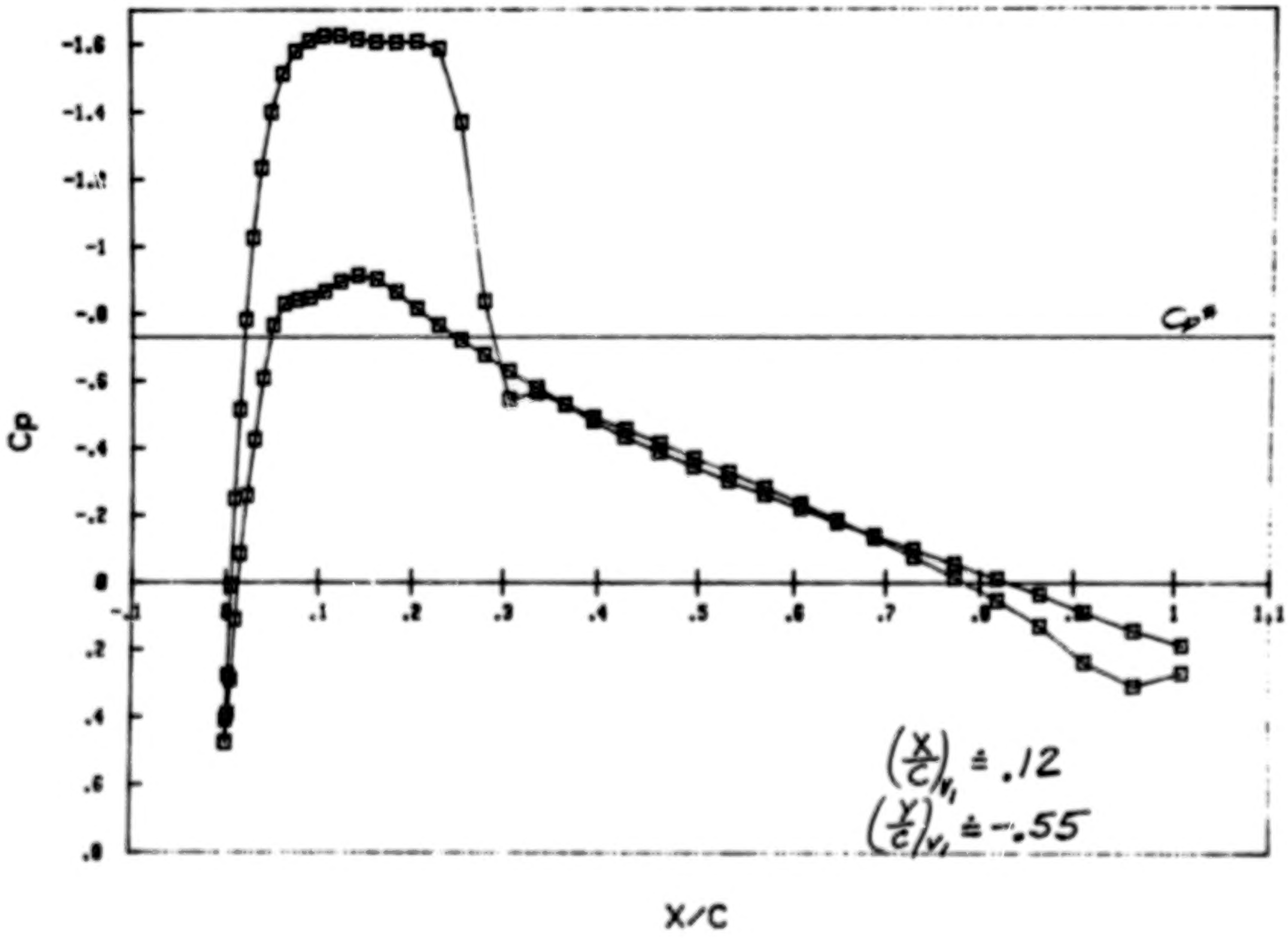


A second vortex, referred to here as the interaction vortex, was placed in the flow field, parallel to the plane of the blade, but at a 10° angle relative to the blade leading edge so that it approached the blade near the tip. This vortex was positioned one-half chord below the blade and 0.13 chord in front of the blade leading edge at the 0.896 span station. Although the vortex accelerates the upper surface flow somewhat in this position, the most pronounced effect is on the lower surface flow which becomes supersonic near the leading edge. The vortex decreases the lower surface pressures so much that the lift is halved at this station. Because a quasi-steady calculation was used to model a transient event, it is likely that the upper limit of the blade/vortex interaction intensity is being illustrated, i.e., the actual flow would lag that shown here.

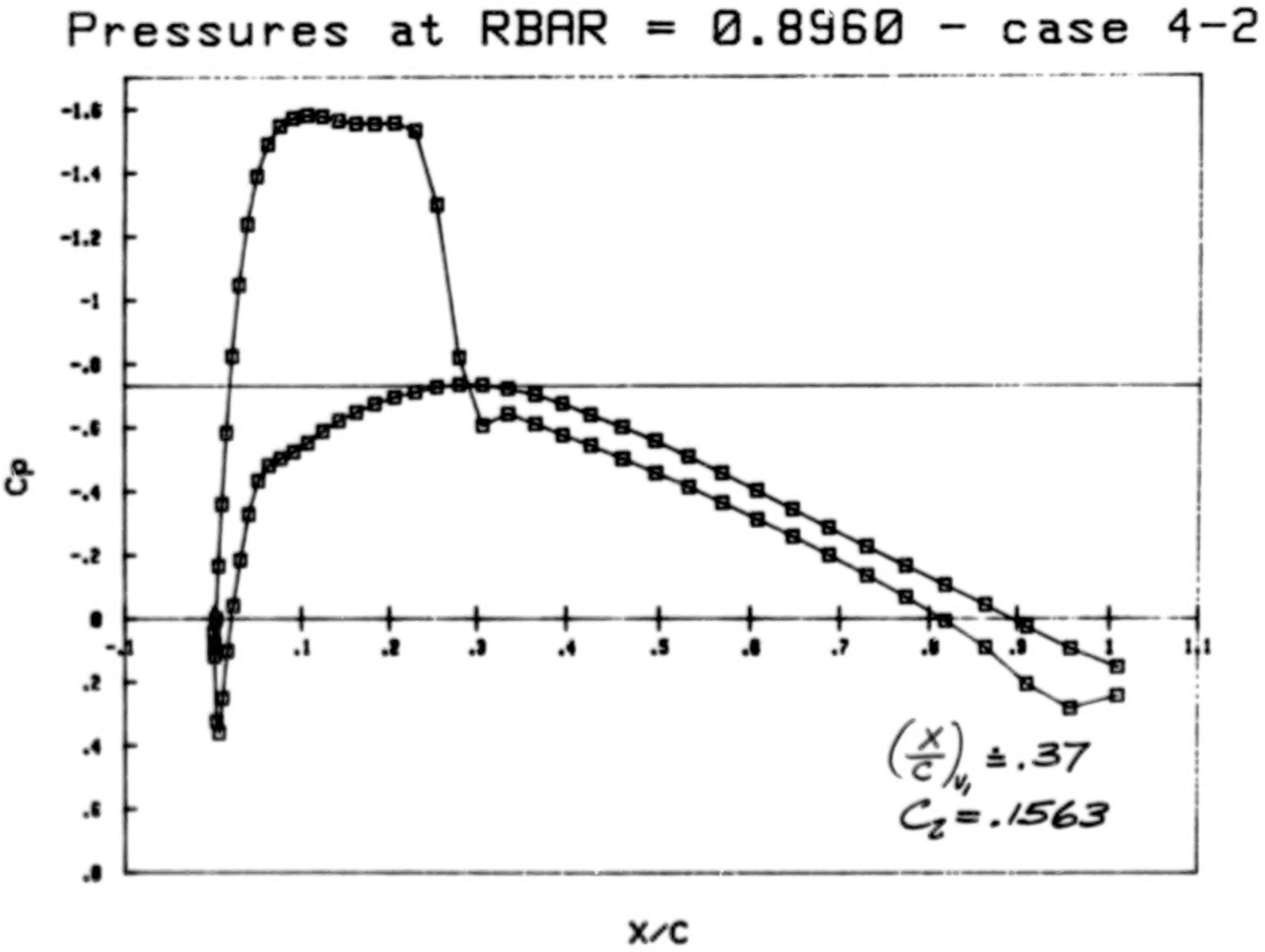


The blade was moved so that the interaction vortex lay 0.12 chord behind the leading edge and 0.55 chord below at the same spanwise blade station. Note that the upper surface supersonic zone, and therefore shock strength has been dramatically increased, while on the lower surface the pressures remain low, but the flow is only slightly supersonic. Thus, in addition to the large loss of lift already noted, the increased extent of the supersonic flow regions and the strengths of the shocks must be considered as major potential sources of noise.

Pressures at RBAR = .896 case 3-2



When the blade was advanced another quarter chord at station 0.896 so that the interaction vortex was located 0.37 chord aft of the leading edge, the upper surface supersonic region remained nearly as large and the shock as strong, but the lower surface pressures increased somewhat. However, the presence of the interaction vortex still reduces the lift at this blade station by over 50 percent.



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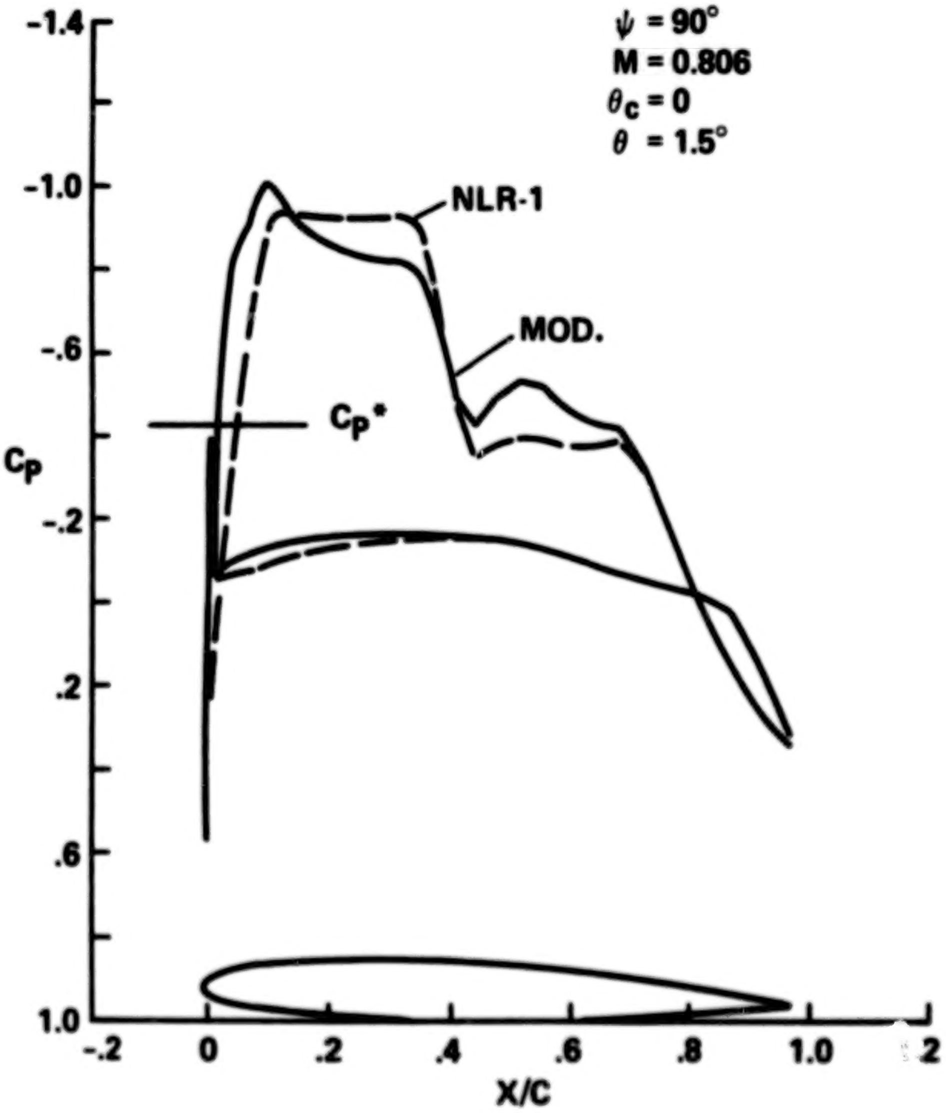
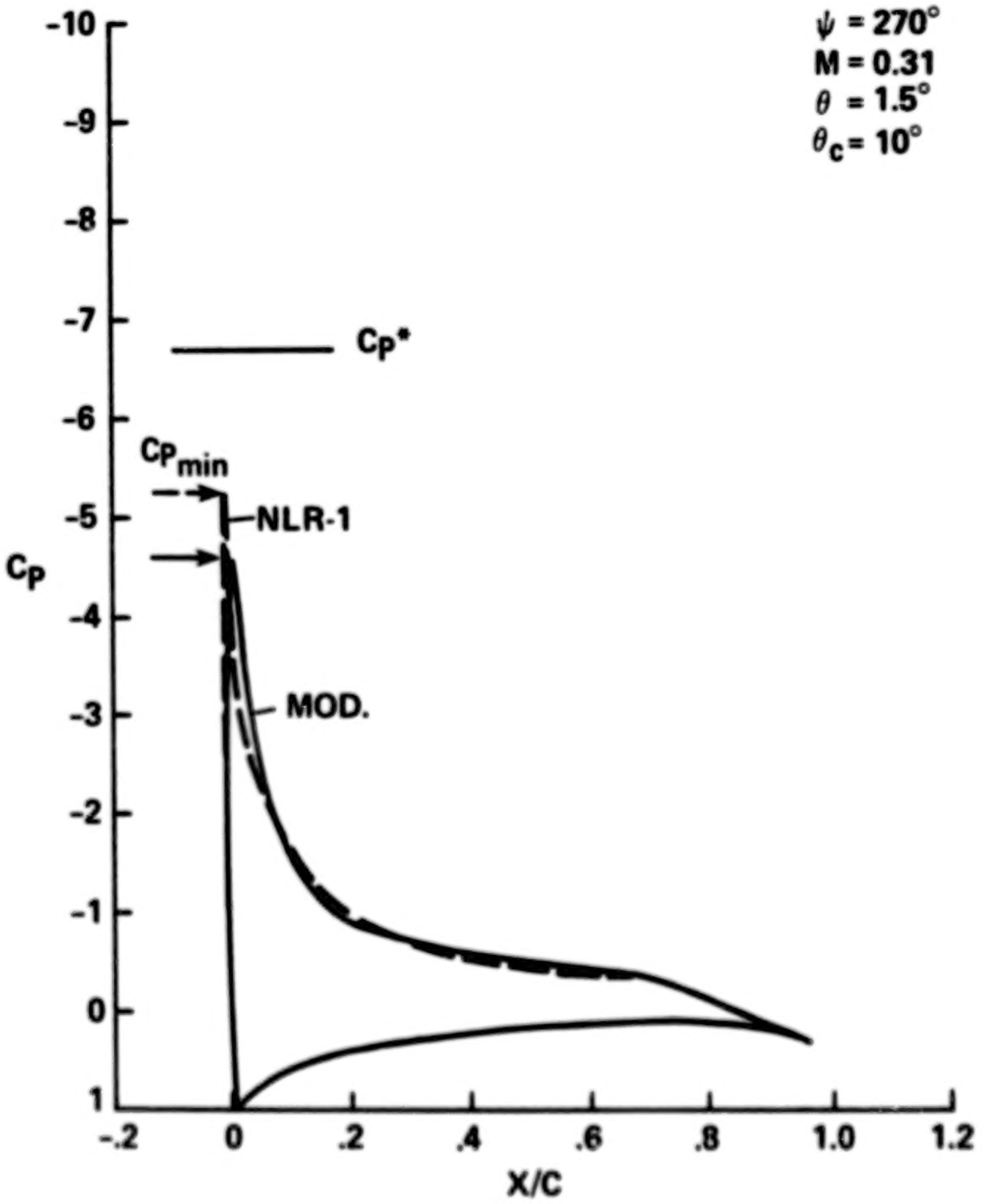
Another mechanism responsible for high power consumption and intense noise is a shock wave forming in the outer regions on the top surface of the advancing blade. In reference 1 it was shown how the ROT22 code could be used to modify the airfoil sections at various radial stations on a hypothetical modern rotor blade which initially had an advanced supercritical airfoil (NLR-1). For an advance ratio of 0.385, corresponding to a tip Mach number of about 0.91, it was shown that reductions in the shock strength on the advancing blade could be achieved, while simultaneously lessening leading-edge pressure gradients on the retreating blade. An example of the shock strength reductions achieved by the airfoil modifications is shown by the solid line; the dashed line represents the pressures on the original NLR-1 section. The example shown is at a radial station of 0.855, where the blade twist was 1.50°. The major blade section modifications required were blunting of the upper surface leading edge and some reshaping of the blade's upper surface resulting in a moderately thicker airfoil.

PRESSURE DISTRIBUTIONS AT r/R = 0.855



While it is relatively easy to modify an airfoil on a blade to improve its transonic performance through reducing the shock strength, it is much more difficult to simultaneously achieve improved retreating blade performance. Since the NLR-1 airfoil experiences leading edge stall under dynamic conditions, it was assumed that reducing the leading edge suction peak and pressure gradient would delay dynamic stall on the retreating blade. Thus, it was shown that a combination of modifications could be made which would both weaken the shock significantly on the advancing side and also reduce the leading edge pressure gradient when the blade is at an azimuth angle of 270° and has 10° of collective pitch.

PRESSURE DISTRIBUTIONS AT r/R = 0.855



In summary, the past and present efforts have consisted of the development, extension, and modification of the three-dimensional ROT22 transonic flow field code. Examples have been shown illustrating the application of the code to studying blade/vortex interaction. It was also shown how a hypothetical modern blade can be redesigned to weaken shock waves on the advancing side while, potentially, improving performance on the retreating side. Future plans include the calculation of acoustic properties using the present and newer, fully unsteady, codes. Initially, computed pressures and velocities will be input into the acoustic propagation codes. However, the eventual goal is to couple the transonic aerodynamic and acoustic codes.

SUMMARY

PRESENT EFFORT

- 3-D ROT22 CODE APPLIED TO
  - STUDY BLADE/VORTEX INTERACTION
  - MODIFY BLADE TO WEAKEN SHOCKS

FUTURE PLANS

- CALCULATE NOISE FROM ABOVE SOURCES
- USE CODE(S) TO DESIGN QUIETER BLADES

REFERENCE

Tauber, M. E.; and Hicks, R. M.: "Computerized Three-Dimensional Aerodynamic Design of a Lifting Rotor Blade," American Helicopter Society Preprint No. 80-2, May 1980.

N84 29682

STATE OF THE ART - DESIGN FOR NOISE I

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OBJECTIVES OF THIS PRESENTATION

This paper is concerned with the engineering management "strategy" involved in designing to meet a noise requirement. Hopefully, this will provide a different perspective on how designs were accomplished in the past, how a new design would be carried out today, and how knowledge we gain through acoustics R&D will have two important effects. It will lead to invention of quieter features to be incorporated, and it will improve our ability to predict accurately the noise levels of new designs before they are built. Each of these effects changes the basic design process.

- **STRATEGY OF THE PAST**
- **TODAY'S NEW-DESIGN APPROACH**
- **A CONFIDENCE - BUILDING PROCESS FOR FUTURE DESIGNS**
- **STEPS NEEDED TO PREPARE FOR THE FUTURE**

Figure 1

PAST DESIGNS CONSIDERED NOISE BUT WEREN'T DRIVEN BY IT

In the past, noise requirements had little effect on design. There were cases where lower tip speeds were chosen, but beyond that, little was compromised. There was no need to. Thinner tips and swept blade tips could help some without performance penalty, but any change that involved compromise with performance was caught between a soft noise requirement and a hard, demanding performance requirement, which carried the day.

This environment hasn't helped the helicopter's image with the public. If we want our production to expand into the large market that could be available, we must reduce objectionable noise whether or not the specifications require it.

- **MAJOR EFFECTS OF TIP SPEED, SIZE, DISC LOADING, POWER WERE KNOWN.**
- **NOISE REQUIREMENTS WERE NONEXISTENT OR LOOSE.**
- **DESIGN TRADE - OFFS PERMITTED LITTLE PENALTY IN THE INTEREST OF LOW NOISE.**
- **SALES OF A PARTICULAR MODEL WERE NOT STRONGLY INFLUENCED, BUT GROWTH OF HELO MARKET SUFFERS.**

Figure 2

REQUIREMENTS HAVE BEEN RELATIVELY LOOSE

To illustrate past requirements, these are Sikorsky's most recent designs. External noise was a concern in the S-76 design. Rules imposed by regulating agencies were really not in place, so a company-selected standard was set.

Military designs have had no requirement until recently. Now detectability concerns have created the beginnings of a set of noise rules there too.

- **CIVIL S-76**
  - **SA:** 90 PNL IN HOVER AT 500 ft
  - **FAA:** ENVIRONMENTAL IMPACT STATEMENT
  - **CAA:** NONE
  - **LOCAL:** RELY ON OPERATIONAL PROCEDURES
- **MILITARY S-61 :** NONE
  - **CH-53A:** NONE
  - **UH-60A:** 1/3 OCTAVE SPEC 1000' AHEAD OF A/C
  - **SH-60B:** NONE
  - **CH-53E:** NONE

Figure 3

INDUSTRY NOW MOTIVATED TO IMPROVE

For many reasons, now, we must face the need to make design compromises to produce a quieter helicopter. The fourth and sixth items on this list aren't always addressed. They are really above and beyond what it takes to meet "rules." The ultimate place of the helicopter in society will depend, in part, on what kind of neighbor it is. Public acceptance will require a good neighbor. For this reason, industry should encourage the military, too, to pursue low noise since in many parts of our country the public perception of helicopters comes from observation of military aircraft engaged in peacetime operations.

- **CIVIL MARKET GROWING**
- **FAA & ICAO WORKING ON RULES**
- **COMMUNITIES SETTING UP RULES**
- **PUBLIC REACTION INFLUENCES MARKET GROWTH**
- **MILITARY DETECTABILITY CONCERNS**
- **MILITARY MUST BE SENSITIVE TO CIVIL CONCERNS**

Figure 4

WITH TODAY'S STATE OF ART, ACOUSTIC DESIGN IS SEMI-EMPIRICAL

If we were starting a new design today, with today's increased awareness of noise, how would we do it? What is today's "strategy of design?" First, we must acknowledge that our ability to predict noise has limitations, and is semi-empirical. If specific goals or rules must be met, we must provide either a margin for error or a means of correcting the end product. Then our methods must be used to assess the trade-offs. How much performance and weight must be given to meet a noise level? To answer questions, tests of existing hardware, perhaps in modified form, may be needed. The new design will fit some niche where our noise data bank is lacking, so tests tailored to the likely final parameters will be run.

Eventually, trade-offs must stop and the design is frozen. Then more detailed analyses can be used if available and model testing can be used to build confidence. If possible, an existing aircraft would be modified to become an acoustic model of the new design. Finally, 18 months after the design freeze, the actual rotor will be whirled and "real" data obtained.

- **SELECT DESIGN GOAL, INCLUDING CONTINGENCY**
- **DEFINE WEIGHT, COST AND PERFORMANCE TRADES**
- **CONDUCT RISK-REDUCTION TESTS**
- **FREEZE DESIGN**
- **DETAILED ANALYSIS AND SIMULATION (TESTING)**
- **FULL SCALE GROUND AND FLIGHT TEST**

Figure 5

C-5

PARAMETRIC MODELS PERMIT DESIGN TRADE OFFS

Backing up, this figure illustrates the preliminary design process. All points on the chart meet the performance requirement. Constraints are placed due to upper limits on disc loading, rotor diameter and noise. Within those constraints, generally, the lightest design is desired. When the parametric relationships between variables (diameter, power, tip speed, etc.) and noise are known, such a chart is quickly generated to permit orderly derivation of a balanced design.

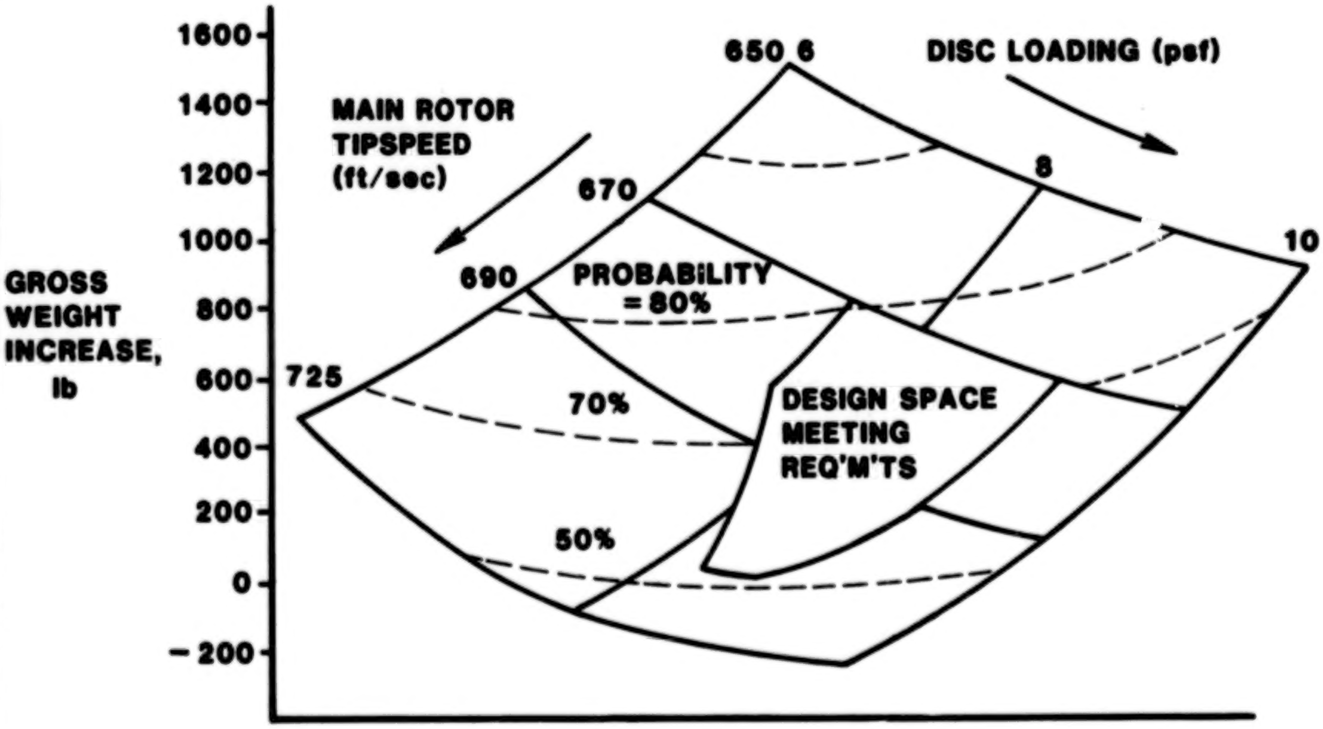

Figure 6

MORE RISK REDUCTION TEST OPTIONS ARE NEEDED

Without a complete, correlated method of predicting helicopter noise, the design process uses all the tools available to provide early answers and reduce risk. On the test side, these include small models, full scale testing of similar design, and finally tests of the design itself. Part of the improved capability we need is obviously in improving these tools. For early answers at moderate costs, we need a calibrated wind tunnel capability. Beyond that, we need to develop and use acoustic measuring facilities for flight test to a much greater extent. Sikorsky is currently developing such a flight test facility at its Florida Development Flight Test Center.

	AVAILABLE	CALIBRATED	COST	TIMING
MODEL SCALE				
• HOVER	YES	YES	LOW	EARLY
• CONV. WIND TUNNEL	YES	NO	MODERATE	EARLY
• ACOUSTIC WIND TUNNEL	LIMITED	IN PROCESS	MODERATE	EARLY
FULL SCALE SIMULATION				
• HOVER	YES	YES	MODERATE	MARGINAL
• FORWARD FLIGHT	LIMITED	YES	HIGH	MARGINAL
FULL SCALE DESIGN				
• HOVER	YES	YES	HIGH	LATE
• WIND TUNNEL	LIMITED	NO	HIGH	LATE
• FORWARD FLIGHT	LIMITED	YES	HIGH	LATE

Figure 7

HELICOPTER HOVER PERFORMANCE PROVIDES A PARALLEL TO THE NOISE PREDICTION PROBLEM

In many ways, the noise prediction problem parallels the rotor hover performance problem that came to light in the middle 60's. Both analysis and test capabilities were found to have shortcomings for the ranges of parameters then being introduced. We launched a major R&D effort, and by the time UTTAS came along, were able to offer both improved performance and accurate predictions. This was accomplished by a mixture of empirical and first-principle analyses backed up by extensive testing. In fact, at the time of the UTTAS proposal there was a heavy reliance on empirical methods and large scale model tests - to predict something as simple as hovering performance. We shouldn't be ashamed if noise predictions take the same route.

- "SIMPLE" CASE PREDICTIONS SERIOUSLY IN ERROR
- REQ'M'TS MORE STRINGENT
- DESIGN PARAMETERS (DISC LOADING , BLADE NO., TWIST) CHANGED
- FUNDAMENTAL RESEARCH ACCELERATED
- SEMI-EMPIRICAL ANALYSES USED UNTIL THEORY AVAILABLE
- MODEL & FULL SCALE RISK REDUCTION TESTS KEY

Figure 8

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FOR HOVER, COMPLEX AERODYNAMIC INTERACTIONS HAD TO BE CONSIDERED IN DESIGN

The reason for the hover performance problem was the complexity and importance of the rotor wake. The photo shows the rapid contraction and the proximity of the wake to the blades themselves. This made performance definition difficult, and it's part of the problem of noise prediction at low speeds as well.

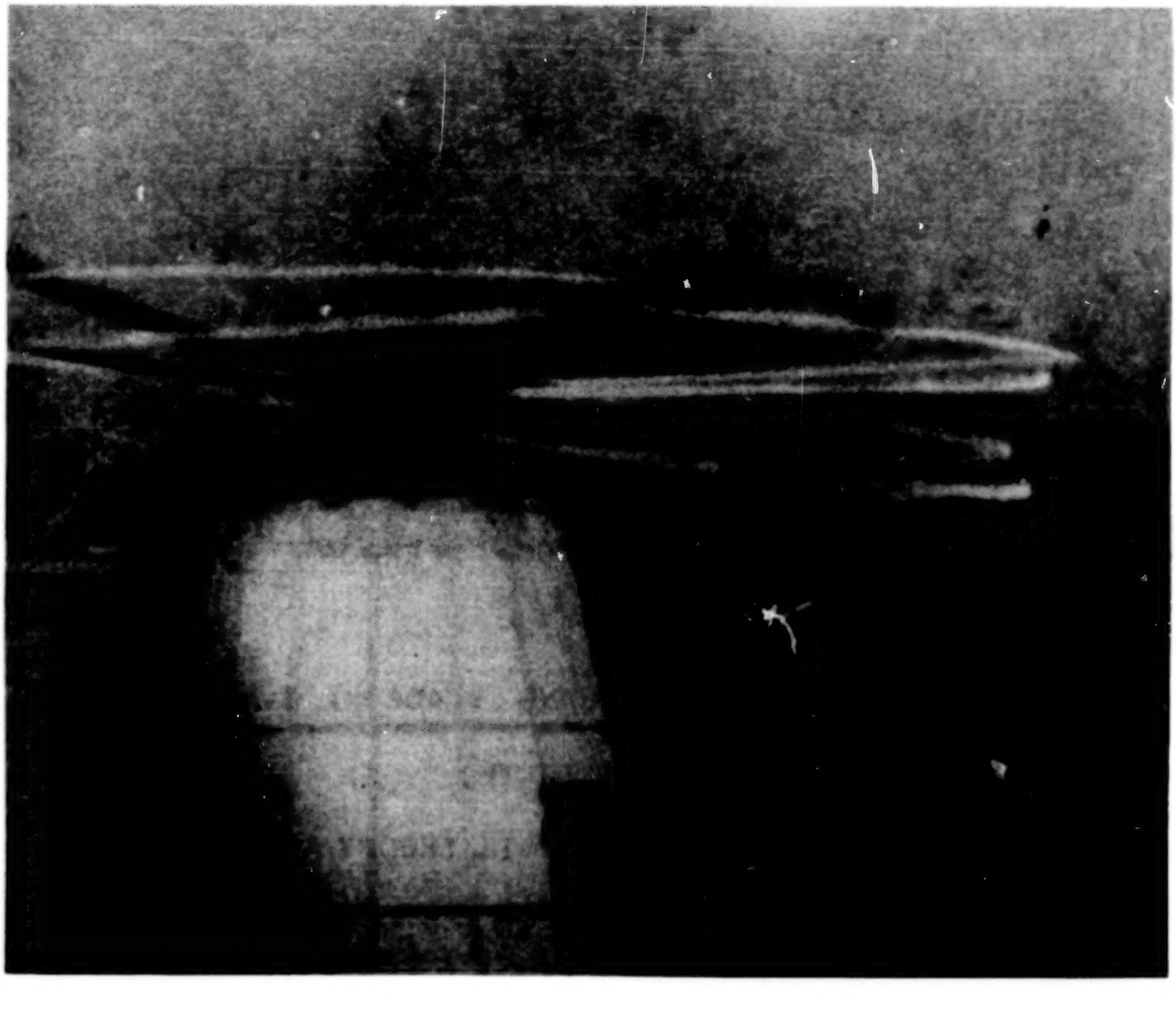

Figure 9

THE "CONFIDENCE PROBLEM"

Once a noise requirement becomes real and has "teeth", the design team must address the risk at failure to meet the requirement, and consider what margins to allow to guarantee success. The weight control analysts in the industry have always used this approach, but it is new to acoustics. Unless a design has unusual flexibility, the die is cast at the design freeze. At that early point, uncertainties may result from limitations of the analyses, changes in the design (such as engine uprating) or inaccuracies of the measurements when the final design is tested. The R&D required today can be viewed as aimed at driving down these uncertainties. In fact, as our understanding of rotor noise generation grows (to drive down these uncertainties), means to reduce noise will become apparent or be invented in the process.

- **RISK OF FAILURE TO MEET REQ'M'TS MUST BE ADDRESSED**
- **FAILURE CAN RESULT FROM**
    - **UNCERTAIN PREDICTIONS (KNOWN & UNKNOWN)**
    - **DESIGN CHANGES**
    - **UNCERTAIN MEASUREMENTS**
- **A JUDGEMENT MUST BE MADE AT DESIGN FREEZE**
- **R&D TASK IS TO REDUCE UNCERTAINTIES**

Figure 10

THE TOTAL UNCERTAINTY IS THE WEIGHTED SUM OF MANY PARTS

The uncertainty is considered to be made up of many parts, and this is an attempt to categorize them. The total is the square root of the sum of the squares, assuming a normal distribution, so some contributors will dominate. Those dominant errors, then, should be subject to the heaviest R&D effort. As gains are made (as in filling out an accurate data base) the emphasis will shift (such as to quantifying the benefits of an exotic tip shape).

- **PREDICTION UNCERTAINTY (σ_p)**
  - **DATA BASE (σ_1)**
  - **SCALING LAWS (σ_2)**
  - **NEW FEATURES (σ_3, σ_4)**
- **MEASUREMENT UNCERTAINTY (σ_m)**
  - **INSTRUMENTATION (σ_5)**
  - **TEST TECHNIQUE (σ_6)**
  - **CONFIG. CONTROL (σ_7)**

TOTAL: $(\sigma_1) = \sqrt{\Sigma \sigma_i^2}$

Figure 11

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R&D PROGRESS CAN BE MEASURED BY REDUCED UNCERTAINTY

If we plot the uncertainty versus time, it should look something like this. Semi-empirical methods are our most accurate tools now, but first principle analyses have the potential for the greatest accuracy, particularly when the design departs in a significant way from past models. Semi-empirical methods will be more accurate for designs that are parametrically similar to past designs. Aerodynamically novel designs, while they may be more quiet, will be less predictable.

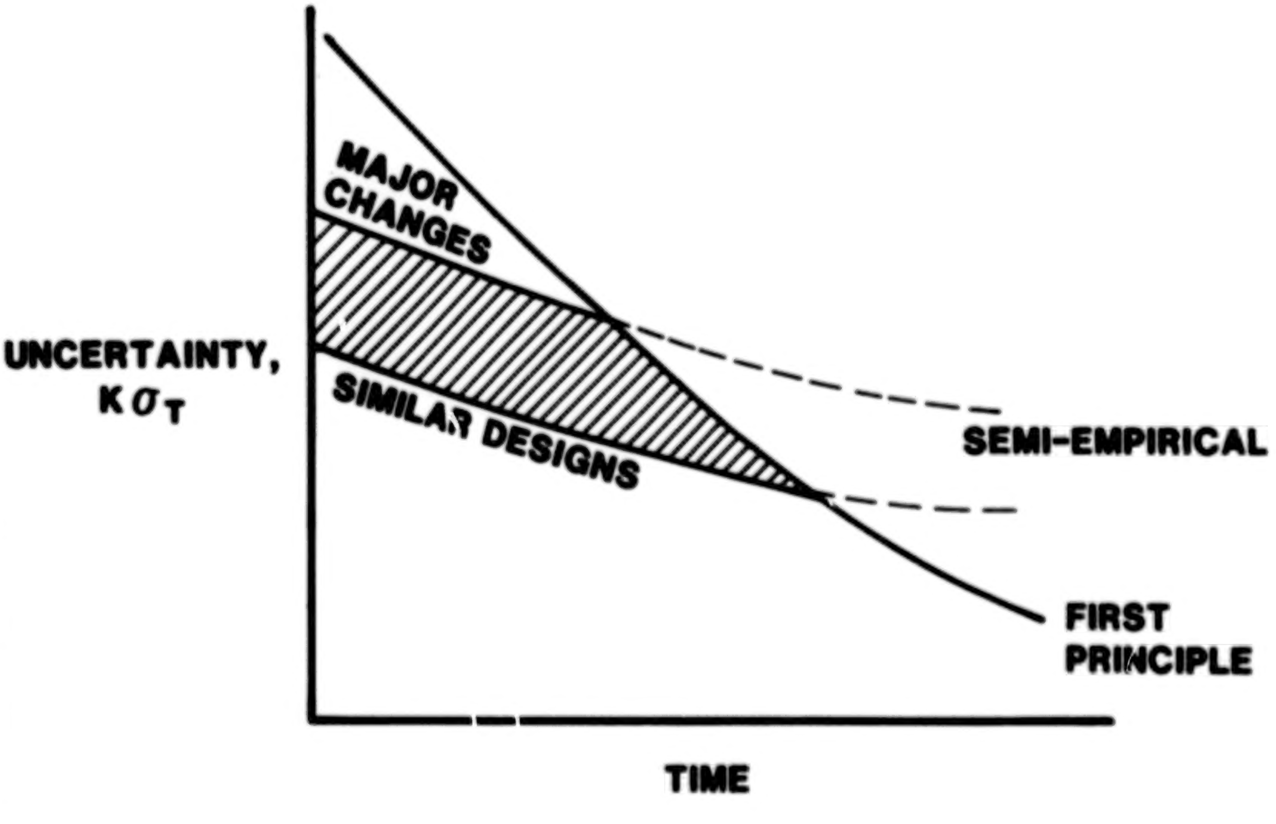

Figure 12

SEMI-EMPIRICAL METHODS CAN FIT MAJOR PARAMETERS

Typically, a semi-empirical noise prediction method will include the variables considered most important and those for which there is a data base. The Sikorsky empirical method, derived from the Ffowcs-Williams-Hawkings formulation of blade loading source, takes into account disc loading, tip speed and forward speed. The scatter of data points, however, indicates that as would be expected, there are other variables that have a measurable effect. In addition, of course, it is likely that EPNdB is related in a much more complex way than the figure implies to even the major variables.

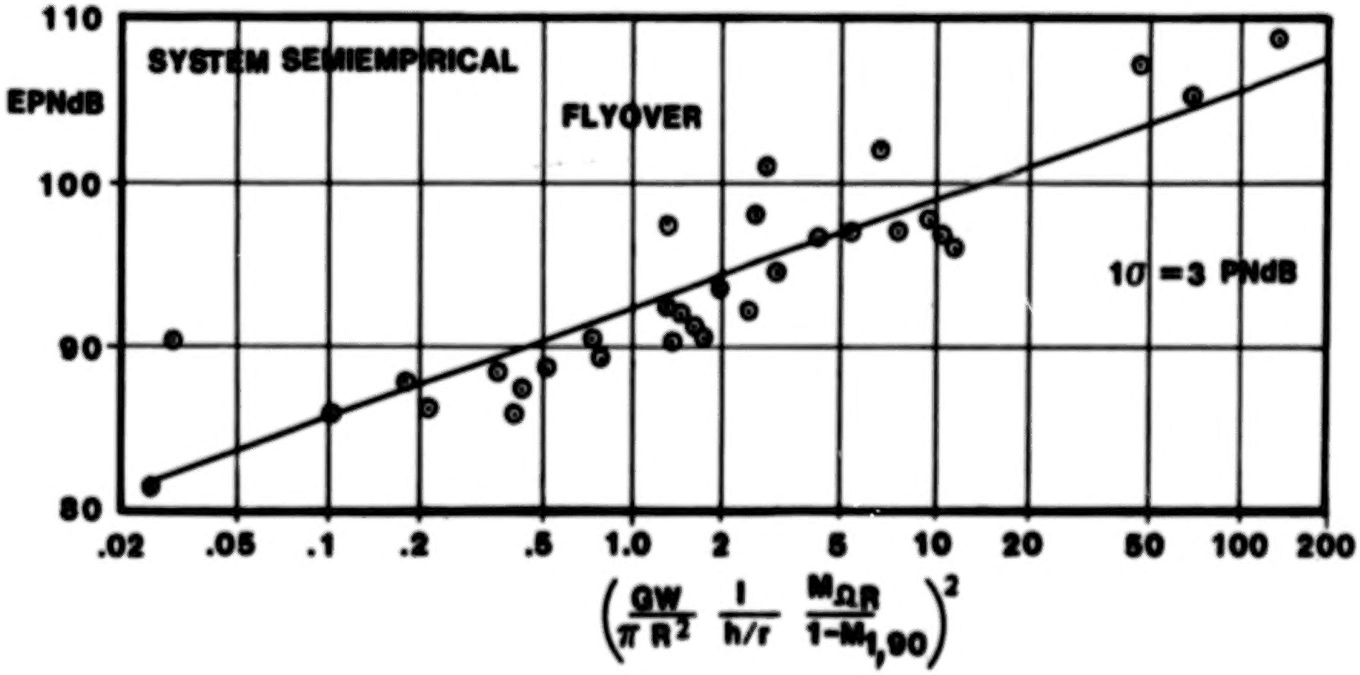

Figure 13

SEMI-EMPIRICAL METHODS ARE SEVERELY CHALLENGED BY POTENTIALLY LARGE NUMBER OF DESIGN PARAMETERS

With sufficient test data, the parametric effects of primary helicopter parameters on noise could eventually be determined, even without a rational theory. However, a wholly empirical solution becomes clearly impractical when some of the more subtle variables are considered. We could never test all combinations of tips, twist and airfoils for example.

Further complication is added to both empirical and theoretical solutions by several potentially important interactions between components. Theory must, at least, help guide the empirical trends to reduce the time and expense involved in understanding the many combinations of effects that are present.

DESIGN FEATURES SUCH AS

- **TIP SHAPE**
- **AIRFOIL**
- **TWIST**

MAIN ROTOR & TAIL ROTOR

INTERACTIONS SUCH AS

- **MAIN ROTOR/TAIL ROTOR**
- **FUSELAGE/MAIN ROTOR**
- **VERTICAL TAIL/TAIL ROTOR**
- **ENGINE EXHAUST/TAIL ROTOR**

Figure 14

R&D THRUSTS NEEDED TO REDUCE UNCERTAINTY

The R&D needed to "solve" the helicopter noise problem is seen as involving five categories of effort. We need to expand significantly the accurate, full-scale data base and, on that basis, to complete a definition of the problem at hand. Test facilities and techniques for both model and full scale tests require development. The validity of model scale testing should be quantified so that time and money may be saved by optimum use of that option. With good test data and procedures, the empirical methods should be refined to produce the best low noise solutions and the best noise predictions possible. In all probability, such methods will always be the appropriate preliminary design tools. Finally, and importantly, the basic understanding of noise producing mechanism must be pursued so that "first principle" analyses can be used to predict trends, aid in invention, and ultimately, minimize the acoustic uncertainty.

- **EXPAND FULL SCALE DATA BASE & UPDATE PROBLEM DEFINITION**
- **REFINE TEST FACILITIES/TECHNIQUES**
- **VALIDATE MODEL TEST CAPABILITY**
- **EXPERIMENTALLY QUANTIFY PARAMETRIC TRENDS AND UPDATE SEMI-EMPIRICAL ANALYSIS**
- **CONTINUE TO DEVELOP FIRST PRINCIPLE ANALYSIS & CORRELATE**

Figure 15

STEP I IMPROVEMENT

For convenience of discussion, the path to a full answer is broken into three sequential steps. Step I, which we should take now, involves five parallel activities with an emphasis on problem definition and a quick pay-off. At the end of Step I we'll have a far better data base and empirical methods based on that data. We'll have improved test capabilities and analyses ready to launch Step II.

- **INCREASE FULL SCALE DATA BASE ON MAJOR SYSTEM DESIGN & OPERATION PARAMETERS**
- **STANDARDIZE WIND TUNNEL TEST FACILITIES/TECHNIQUES**
- **CONDUCT QUALITATIVE MODEL SCALE PARAMETRIC TESTS**
- **MAKE AVAILABLE A MK I 1st PRINCIPLE ANALYSIS FOR EVALUATION**
- **DEVELOP MK II SEMI-EMPIRICAL ANALYSIS**

RESULT: DESIGN MARGIN REDUCED FROM 5-6 dB TO 4-5 dB

Figure 16

STEP II IMPROVEMENT

In Step II the experimental work can progress beyond baseline data to quantify the effects of more subtle design variables, such as airfoils or tip shape or rotor spacing on noise. Both model and full scale facilities should contribute at the point. At the same time, both the data and the analyses will be in hand for evaluation. A significant activity in Step I, by the way, should be the prioritization of analytical developments needed to design "neighborly" helicopters. At each step thereafter, we need to update those priorities.

- **INCREASE FULL SCALE DATA BASE ON DETAIL DESIGN PARAMETERS/INTERACTIONS**
- **VALIDATE MODEL TEST CAPABILITY**
- **DEVELOP MARK III SEMI-EMPIRICAL ANALYSES**
- **MAKE AVAILABLE MK II FIRST PRINCIPLE ANALYSIS FOR PARAMETRIC TRENDS**

RESULT: DESIGN MARGIN REDUCED FROM 4-5 dB TO 3-4 dB

Figure 17

STEP III IMPROVEMENT

The final R&D step provides the balance of the necessary tools - a full data bank, accurate test techniques and corrrelated analyses to handle the infinite variety of designs that industry can create. Little has been said, in describing these R&D steps, about invention of low noise solutions. My assumption is that such inventions will happen automatically as our knowledge of noise sources grows. Low noise solutions will pay off in smaller design penalties to meet specific goals, so the incentive is always there. This plan is described in terms of knowledge building rather than inventing, as an intentional over-emphasis. A certain amount of cut-and-try is healthy, but the long term goal requires that we do the homework.

- **DEVELOP EXTENSIVE DATA BASE ON INTERACTION EFFECTS**
- **VALIDATE MK III FIRST PRINCIPLE ANALYSIS FOR ABSOLUTE PREDICTION**
- **REDUCE TEST UNCERTAINTIES**

RESULT: DESIGN MARGIN REDUCED FROM 3-4 dB TO 1.5-2.5 dB

Figure 18

PROJECTED REDUCTION IN UNCERTAINTY

If each of the R&D steps takes two years, the reduction of acoustic uncertainty would look like this. It is estimated that with today's tools, providing a design margin to give a high confidence of meeting noise requirements could increase the aircraft size and raise operating costs by 20 to 40%. By the end of Step III, it should be possible to cut the uncertainty by 60% and reduce the DOC impact of noise prediction below a measurable threshold. This does not say that meeting noise rules will cause no penalty, but only that no added penalty must be imposed due to imperfect analyses. The absolute penalty of noise requirements will depend, of course, on the level of the requirements themselves. A significant output of the R&D effort will be proper evaluation of the implied cost of noise control. As we collectively get smarter about what is possible in design, we can work for noise requirements that properly balance the user or the public desires against the economic impact of noise control.

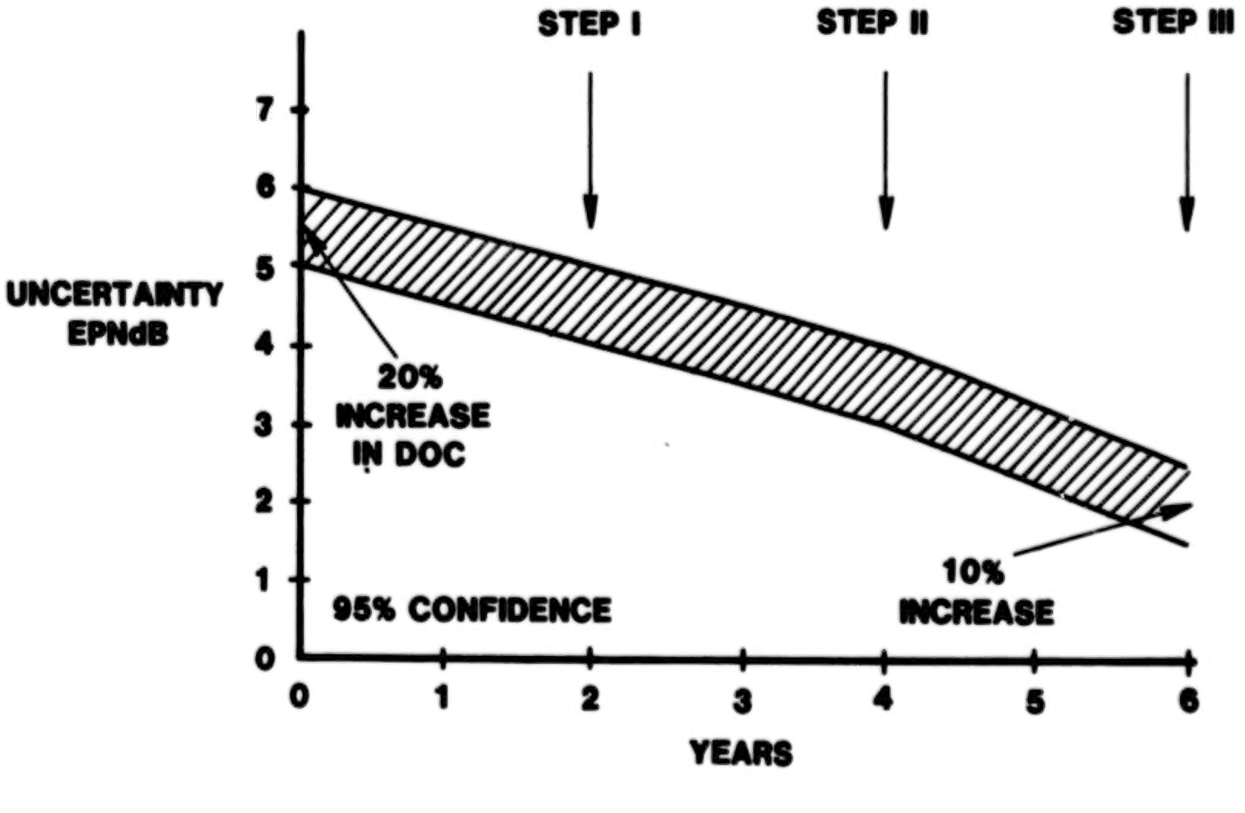

Figure 19

CONCLUSIONS

To sum up - we see helicopter acoustics R&D as a continuing blend of analytical and experimental developments. The progress can be tracked in terms of the reductions achieved in the acoustic uncertainty of new designs. The time scale will depend on both the level of effort and the cleverness of the researchers. The expansion of the helicopter industry to its proper place in the world awaits the successful completion of such a program.

- ANALYSIS AND TEST MUST PROGESS TOGETHER
- PROGRESS WILL INITIALLY BE MADE BY SEMI-EMPIRICAL MEANS AND CARRIED FORWARD BY FIRST-PRINCIPLE ANALYSES
- MILESTONES SHOULD BE SET AND PROGRESS MEASURED IN TERMS OF REDUCTION IN UNCERTAINTIES
- THE METHODOLOGY, BOTH DESIGN AND TEST, WHICH SHARPENS PREDICTIONS WILL LEAD TO NEEDED NOISE REDUCTION

Figure 20

N84 29683

STATE OF THE ART - DESIGN FOR NOISE II

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Because this paper was not available at time of publication, only slides are presented.

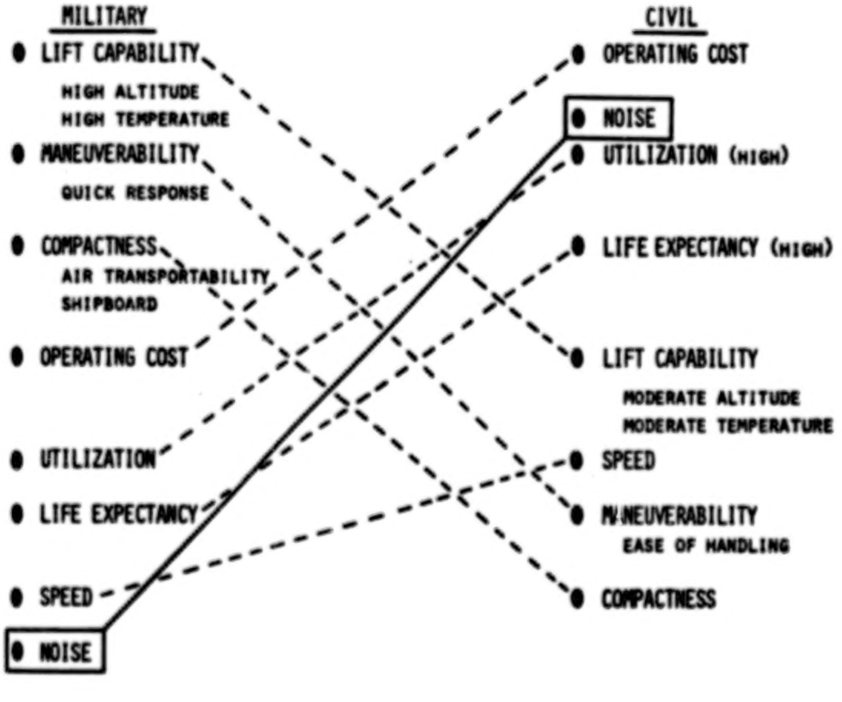
DESIGN FOR NOISE

→ APPROACH

CRITERIA & REQUIREMENTS

CASE HISTORY

PRIORITIES



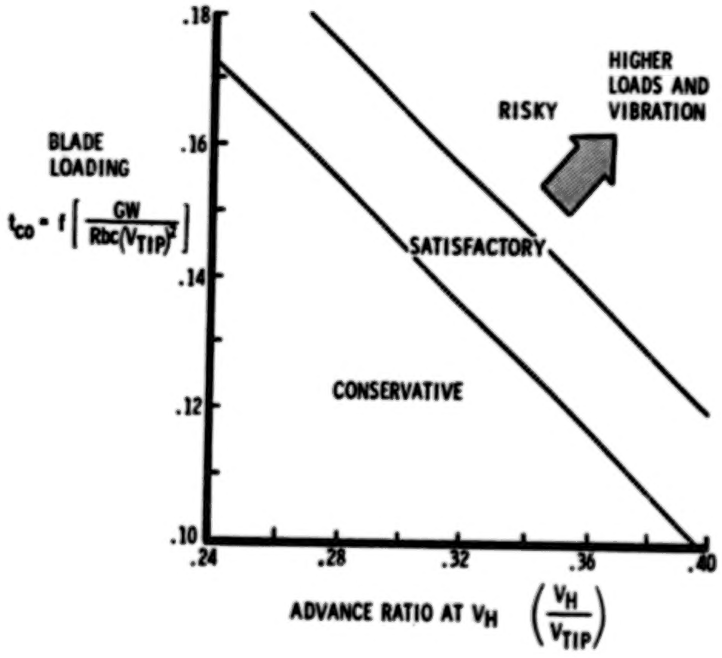
MILITARY

- LIFT CAPABILITY
  - HIGH ALTITUDE
  - HIGH TEMPERATURE
- MANEUVERABILITY
  - QUICK RESPONSE
- COMPACTNESS
  - AIR TRANSPORTABILITY
  - SHIPBOARD
- OPERATING COST
- UTILIZATION
- LIFE EXPECTANCY
- SPEED
- NOISE

CIVIL

- OPERATING COST
- NOISE
- UTILIZATION (HIGH)
- LIFE EXPECTANCY (HIGH)
- LIFT CAPABILITY
  - MODERATE ALTITUDE
  - MODERATE TEMPERATURE
- SPEED
- MANEUVERABILITY
  - EASE OF HANDLING
- COMPACTNESS

DESIGN GUIDELINES



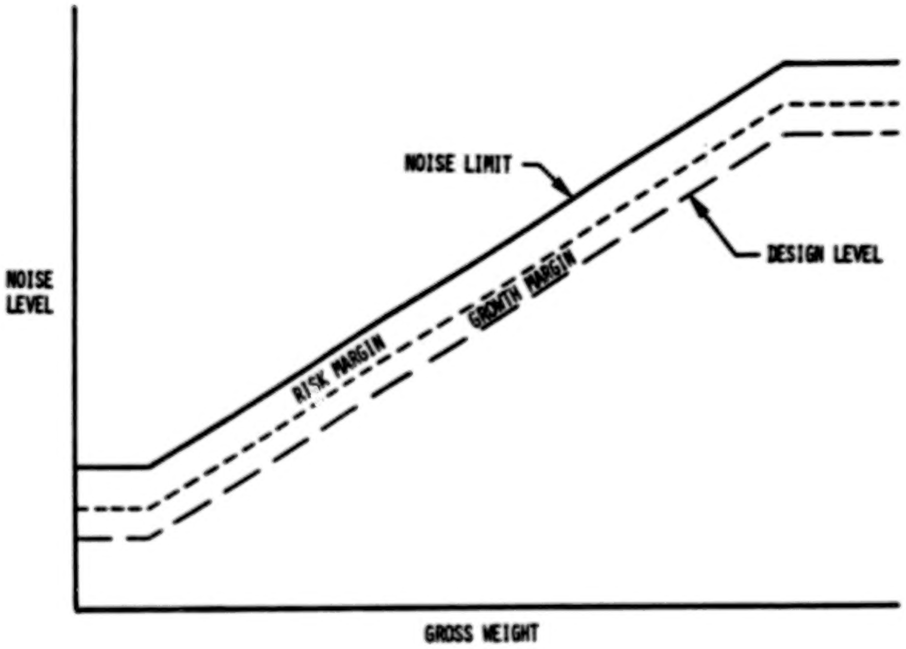
DESIGN FOR NOISE

APPROACH

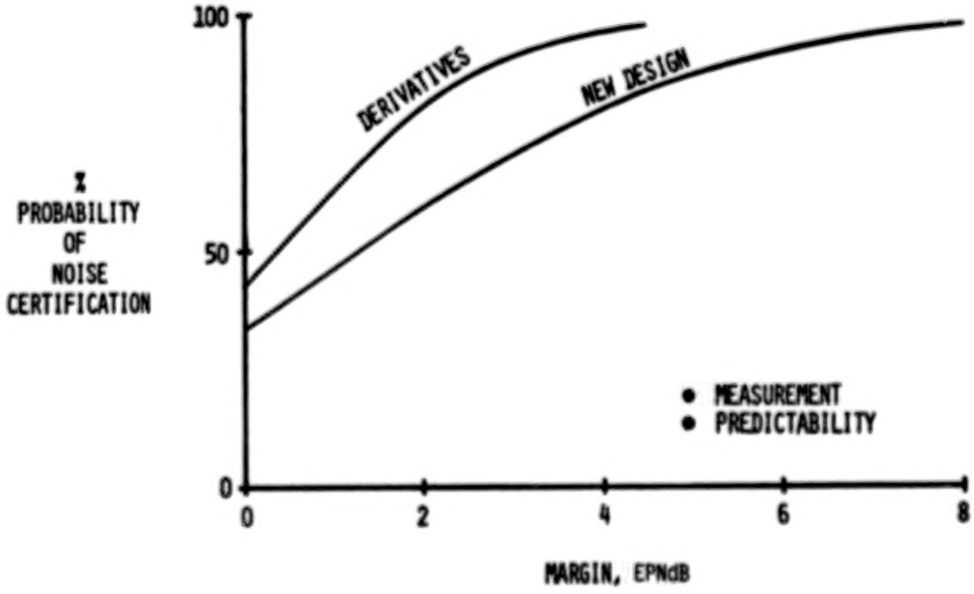
→ CRITERIA & REQUIREMENTS

CASE HISTORY

DESIGN MARGINS



RISKS



NOISE PREDICTION

CONDITION	ACCURACY
FLYOVER	
- MODERATE AIRSPEEDS	± 2 EPNdB
- HIGH AIRSPEEDS	± 3 EPNdB (AT BEST)
	± 5 TO 8 EPNdB (SOME CONFIGURATIONS)
APPROACH	
- WITH WAKE INTERACTION	NO PREDICTION METHOD
TAKEOFF	NO PREDICTION METHOD

NOTE: NO GENERALLY ACCEPTED PREDICTION METHODS EXIST FOR THE VARIOUS TYPES OF ANTI-TORQUE DEVICES.

DERIVATIVE TRENDS

- A DERIVATIVE EVOLVES EVERY 3 TO 5 YEARS
- EACH DERIVATIVE GROWS IN SIZE AND CAPABILITY
- THE CABIN IS ENLARGED EVERY 10 YEARS
- AN ALL NEW DRIVE SYSTEM IS INCORPORATED EVERY 10 YEARS
- FOR EACH DERIVATIVE:

GROSS WEIGHT INCREASES	10 TO 20%
CRUISE AIRSPEED INCREASES	7 TO 14 KTS
M/R MACH (ADV. TIP) INCREASES	1.5 TO 3%
M/R DISC LOADING INCREASES	7 TO 12%

DESIGN FOR NOISE

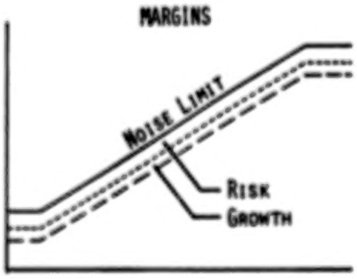
APPROACH

CRITERIA & REQUIREMENTS

→ CASE HISTORY

GUIDELINES: DERIVATIVE HELICOPTERS

- MODEL SERIES CONSIDERED OVER 20 YEAR PERIOD
- DERIVATIVE EVOLVES EVERY 5 YEARS
  - NEAR TERM DERIVATIVES REFLECT ACTUAL MANUFACTURER PLANS
  - FAR TERM DERIVATIVES GUIDED BY PAST HISTORY
- DERIVATIVES MUST BE DESIGNED TO MEET NEW DESIGN REQUIREMENTS
- IN DESIGNING TO MEET NOISE STANDARDS, MARGINS MUST BE INCLUDED (APPLICABLE TO EACH SEPARATE FLIGHT CONDITION)



	MARGIN* (EPNdB)		
	RISK	GROWTH	TOTAL
NEW DESIGN	4.0	1.5	5.5
DERIVATIVE	2.0	1.5	3.5

*ICAO 4/3 TRADEOFF ALLOWANCE INCLUDED

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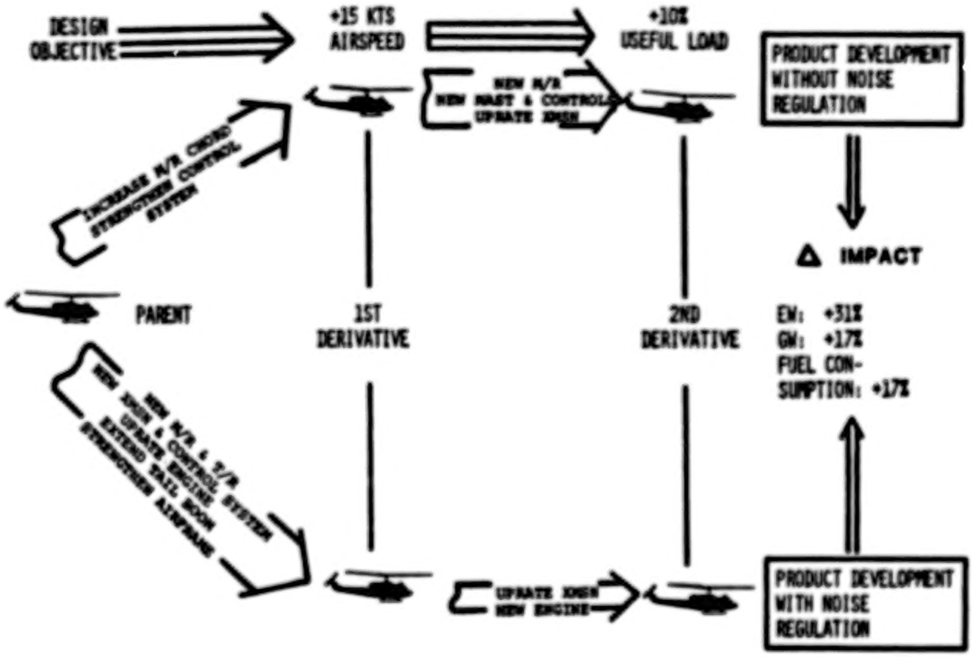
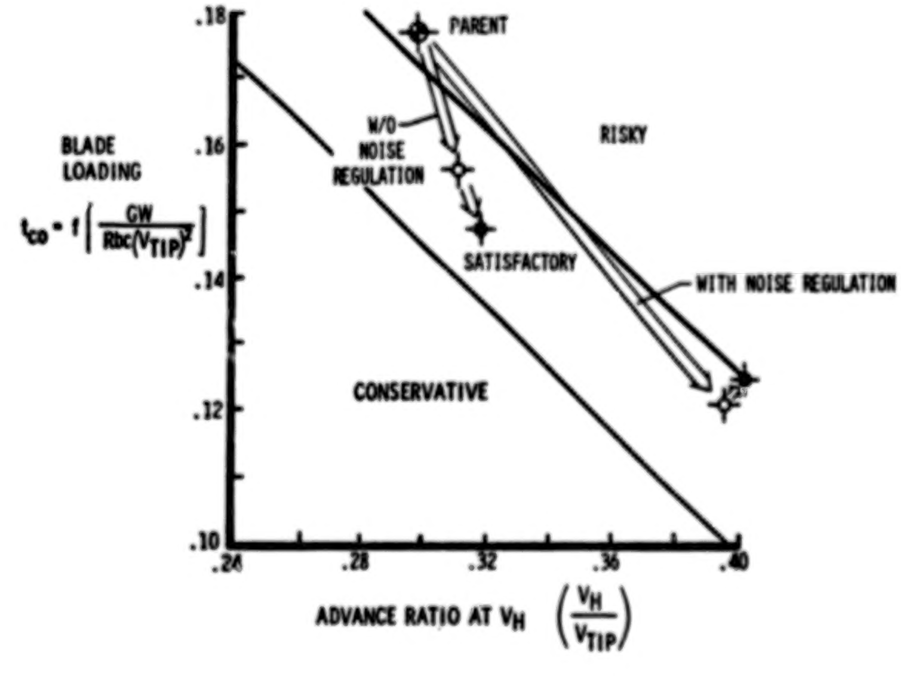
DESIGN IMPACT


DESIGN OBJECTIVE
+15 KTS AIRSPEED
+10% USEFUL LOAD
PRODUCT DEVELOPMENT WITHOUT NOISE REGULATION
NEW M/R
NEW MAST & CONTROLS
UPRATE XMSN
INCREASE M/R CHORD
STRENGTHEN CONTROL SYSTEM
Δ IMPACT
PARENT
1ST DERIVATIVE
2ND DERIVATIVE
EW: +31%
GW: +17%
FUEL CONSUMPTION: +17%
NEW M/R & T/R
NEW XMSN & CONTROL SYSTEM
UPRATE ENGINE
EXTEND TAIL BOOM
STRENGTHEN AIRFRAME
UPRATE XMSN
NEW ENGINE
PRODUCT DEVELOPMENT WITH NOISE REGULATION

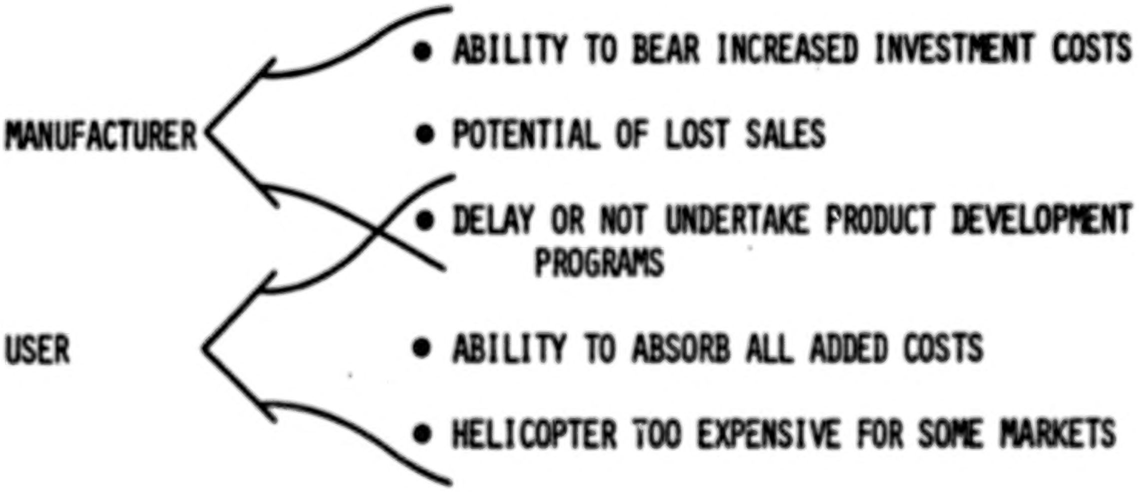
DERIVATIVE DESIGN



COST IMPACT

	DIFFERENCE BETWEEN DERIVATIVE DESIGNS WITH AND WITHOUT NOISE REGULATION
EQUIPPED PRICE	+ 33%
COST PER SEAT MILE	+ 18%
COST OF REGULATION PER AIRCRAFT PER YEAR	+ $.58 M

OTHER IMPACTS



MANUFACTURER

USER

- ABILITY TO BEAR INCREASED INVESTMENT COSTS
- POTENTIAL OF LOST SALES
- DELAY OR NOT UNDERTAKE PRODUCT DEVELOPMENT PROGRAMS
- ABILITY TO ABSORB ALL ADDED COSTS
- HELICOPTER TOO EXPENSIVE FOR SOME MARKETS

N84 29684

A DESIGNER'S VIEWPOINT - REQUIREMENTS FOR REDUCING HELICOPTER NOISE

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Culver City, California

9 207

ABSTRACT

. . . present, from chief designer's point of view the functional requirements of a complete design-for-noise capability . . .

As I reviewed the abstract of my planned discussion I realized it was abstract enough that it covered anything I chose to say. Accordingly, I am going to discuss "Functional Requirements" from the work environment point of view, i.e., what conditions must exist before the designer can really take aim, without too many restraints, at reducing noise. In addition, I'm going to spend a little time on the technical aspects of noise reduction, based on Hughes experience, and offer some thoughts on future design.

Being last on the agenda, following such a formidable array of experts, is both good news and bad news, depending on one's point of view. The good news is that there is not much more to listen to; the bad news is that if I make some of you angry, and I might, I can get away before you shoot back.

I think that the general specifications that follow are mandatory to provide the designer with the complete functional capability to do this job.

1. Community Demand for Quiet

2. Regulatory Requirements

3. Competition

4. Commitment

5. Challenge to the Technical Community

6. Acceptability of Penalties for Reduced Noise

7. Imagination and Skill

I will discuss each of these requirements separately.

COMMUNITY DEMAND FOR QUIET

The sign, "QUIET," is generally associated with hospitals, morgues, and other forboding places. "THE SILENCE IS DEAFENING" is a graphic description of how we react to a change from our noisy world. Our society almost demands noise, or it gets edgy, upset or concerned about some impending disaster. In many urban areas, the best defense against boredom and poor living conditions is a walking stereo system at about 100 dB. That system is now being replaced by lightweight earphones, blasting stereo into the ears of those unknowingly engaged in slowly losing their hearing. Rock music has to be loud. Applause has to be deafening. Making the most noise possible at concerts, ball games and at the theater are signs of approval, and on, and on.

So, does the community want quiet and, most importantly, is it willing to pay for it? As a personal point of view, I do not think the community really objects to the noise, I think the objection is to the "danger" the noise implies. After all, in Los Angeles, the residents who had their property values drop because of proximity to the airport learned to modify the noise with drapes and shutters and competitive noise inside the house, and those homes are still desirable and the property values are almost back to par.

So, although I am not sure that the community really demands quiet, I will assume that they really do, otherwise there would be no point in my being here presenting this paper.

REGULATORY REQUIREMENTS

When and if this community demands quieter helicopters, it seems to me that we will need regulatory requirements which would provide minimum noise standards. For a short period of time, when I was a temporary member of the HAI noise committee I found that my opinion was in the minority. The FAA's proposed requirements, in the recent aborted attempt to issue regulations, were irrational with regard to current production. I did agree with the committee on that point. However, I was dismayed that the reaction to the proposed noise standard for helicopters of the future, as arbitrary as it was, was opposed so vigorously. The committee gave lip service to:

"We are not getting enough time"

"We cannot predict well enough"

"The requirements are too severe"

"We need R&D money"

I believe that what was at the heart of the matter was a basic unwillingness on the part of the technical community to agree that the need was important.

"Americans do not really want small cars," and "We do not know how to reduce the emission levels in such a short time," were the rallying points about which automotive industry almost committed hara-kari. If that industry's managers had not been so concerned with this year's "bottom line" and had exercised a little futurism, we might still be the world's unchallenged leader in automobile production.

It is my personal judgement, at least for small helicopters, that to attain the maximum in growth and sales potential, we must make helicopters "noiseless" and "vibrationless." If the community really wants quiet, then the recent withdrawal of the proposed noise regulations was throwing the baby out with the bathwater. Sensible regulatory action on the part of the FAA, without the punitive aspects of the last proposal, is badly needed and might serve as a welcome catalyst for future designs.

COMPETITION

The establishment of regulatory guidelines, and the acceptance by the major forces in the industry that reduced noise requirements could increase sales, would spur competition among companies and among engineers to really get going. Without competition, the industry would not have made the dramatic gains in weight fractions, reliability and cost of operation that marked the last 30 years. The competition I am talking about is not for this year's helicopter, or for next year's derivative. I am talking about helicopters for the twenty-first century. For those helicopters, minimum noise weight allowances, external and internal, should be set for new designs and should become as important as all the other designer's goals.

This is my 43rd year in aviation, almost all of it in helicopters. My background is varied. However, as a designer, I have never had the challenge of mandatory noise goals. Designers would welcome the challenge. I know I'm ready.

COMMITMENT

Commitment has to start from the top. If and when a company's management determines that sales will be improved by making helicopters quieter, or when the specifications for a competition include noise as a prime criterion, company commitment will become a reality. Then and only then will the doers in the company have to become serious. Perhaps then we will stop mouthing platitudes about what we do not know. They will not be acceptable in a competitive environment in which chief executives become demanding. The commitment to noise reduction will become significant only if the parametric considerations for the next generation of helicopters list noise near the top of the list.

CHALLENGE TO THE TECHNICAL COMMUNITY

Noise experts do not get much attention in engineering divisions. They are not taken very seriously. When I was a noise expert (only temporarily as a member of the HAI Acoustics Committee) our company, in some of its proposal efforts, was suggesting helicopters which would not meet the FAA's proposed requirements. When I protested vigorously, I was told that when the regulations became fact, we would then worry about how to meet them.

Perhaps acousticians are not taken seriously because they take themselves too seriously. In a pseudo-science, where the basic consideration is the subjective reaction of the listener, we are still haggling about which descriptor is most accurate. When I recently suggested that one of our experts should be more demanding for the inclusion of his point of view in rotor system design and weight fractions, his unspoken answer was, "What's the use. No one pays much attention." Now that I think about it, maybe I should have said "Because they do not take themselves seriously."

When the technical community in noise reduction becomes insistent that they be listened to, when individuals in companies become noisy and insistent that they are important, then perhaps they will get some attention. And more importantly, perhaps noise experts will move from staff positions into line positions with design responsibility. Perhaps this may already be true in isolated cases, but I do not think it is the general rule.

ACCEPTABILITY OF PENALTIES FOR REDUCED NOISE

How, in the past and even now, have designers dealt with noise? The alternatives were:

- forget it
- absorb it
- cancel it
- mask it
- do not produce it

"Forget it" has been the primary technique of the past. No one has asked much of designers, so we have conveniently forgotten noise. Once we get past "forget it," we have preoccupied ourselves with absorbing or muffling it, and we immediately equated noise reduction with inefficiency, i.e., unnecessary pounds, decreased performance and increased cost.

Some attempts have been made at canceling. It works well in the laboratory and there has been some progress in noise-canceling headsets. Past that, I think we have "canceled" noise canceling as a possibility. A very useful technique has been, at least for internal noise, to mask it by the creation of more pleasant competitive noise. We can do that fairly well with music, like stereo headsets for passengers. Airlines realized that very early in the game. However, I would like to direct my attention to external noise and techniques for not producing it. Under the general heading of not producing it, I can include as alternatives:

Eliminate flying

Reduced rotational speeds

Reduced tip speeds

Aerodynamic refinements

Non-responsive structure

Reduced propulsion noise

I think it is clear that aside from not flying, reduced rotational speeds and reduced tip speeds are the most powerful. Are the penalties for such reductions debilitating in a competitive environment? I do not think so.

To date, Hughes has twice obtained FAA certification for derivative designs, in response to user demands, for quiet versions of an existing helicopter. In 1973, in a community in Southern California using our Model 300 helicopter for police patrol, there was so much fuss raised about the flyover and orbit noise that the police department was faced with the loss of their helicopters. They asked for help, fast.

We certificated the Hughes 300CQ. It was a basic 300, operating at reduced rotor speeds at specific conditions of minimum altitude and minimum forward speed. Also we added an extra muffler and pointed the exhaust upward. Although the dBA reduction was significant, approximately 6 dBA (see Table I), even more significant from the outer point of view was the difference in recognition distance (fig. 1), which was much greater than the dBA reduction implied. What was the penalty? Only 3 feet of 2 inch tailpipe and a small muffler. Hardly enough to mention.

TABLE I. COMPARISON OF NOISE LEVELS 300 AND 300CQ - INGLEWOOD POLICE DEPARTMENT, FEBRUARY 1973

Flight Condition	dBA	
	300C	300CQ
Ground Warmup	73	68
Hover	91	84
300 ft Flyover	76	69
500 ft Flyover	72	66
700 ft Flyover	68	64
500 ft Orbit	70	65

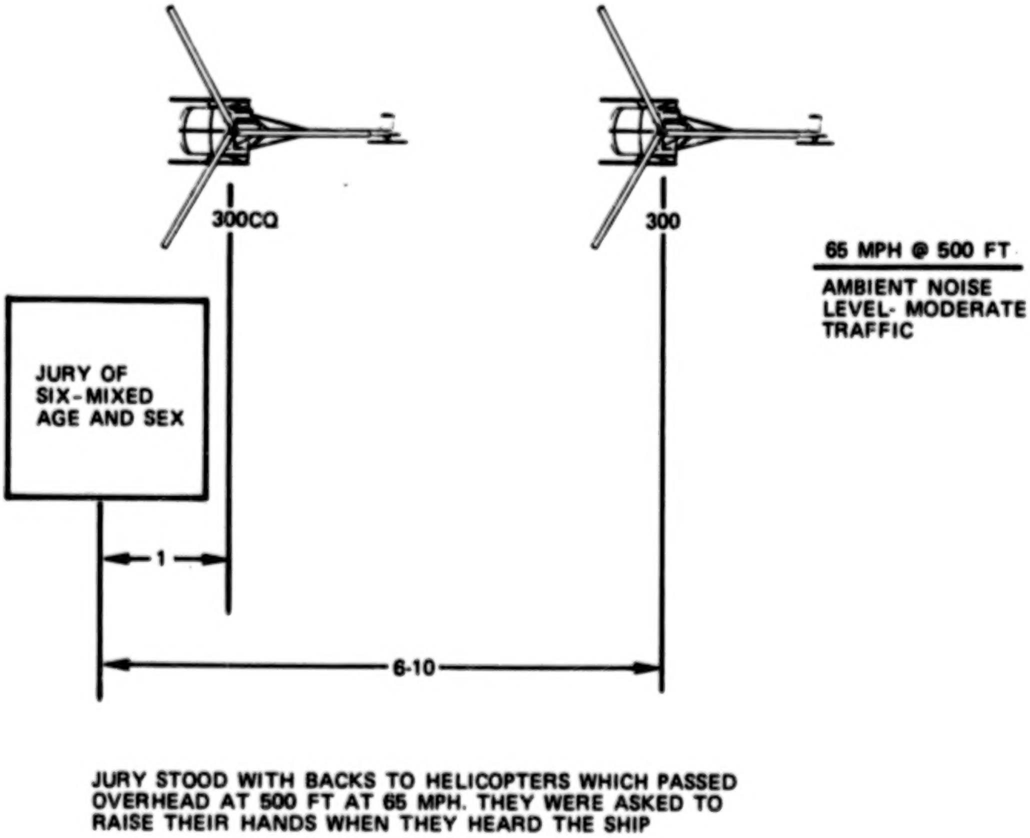
300CQ
300
65 MPH @ 500 FT
AMBIENT NOISE LEVEL- MODERATE TRAFFIC
JURY OF SIX-MIXED AGE AND SEX
1
6-10
JURY STOOD WITH BACKS TO HELICOPTERS WHICH PASSED OVERHEAD AT 500 FT AT 65 MPH. THEY WERE ASKED TO RAISE THEIR HANDS WHEN THEY HEARD THE SHIP

Figure 1. Recognition Distance Comparison Model 300 and 300CQ Using A Subjective Jury.

Three years ago we certificated the Quiet 500. We added two additional tail rotor blades and reduced the tip speed from 704 to 530 fps. Nothing was done to the main rotor. The reduction in the tail rotor noise was approximately 6 dBA. However, the overall noise level was reduced by only 2 to 3. Once again, the reduction in recognition distance was 6 to 10 times.

The penalty was approximately a 1 percent increase in gross weight and 2 percent in cost. For the user who needed quiet, the penalty was insignificant. Although both the 300CQ and Quiet 500 were special cases, perhaps our concepts about how much noise will cost the user have to be re-examined objectively when we think about twenty-first century helicopters.

In each of these cases the difference in recognition distance was measured by a jury of six, mixed ages, men and women. They stood with their backs to helicopters of both types, during a flyover, and were asked to raise their hands when they heard the approaching helicopter. Stopwatch timing was used to calculate distances. Not too scientific, but very revealing. Incidentally, when the experiment was repeated with the group watching the approaching helicopter, they "heard" it sooner.

Several years ago we did a study for NASA in which we compared three arbitrarily chosen planform shapes as replacements for the existing main rotor blade on the Hughes Model 500C helicopter. The purpose was to check the impact of these different planforms (fig. 2) on performance, including noise. The results shown in Table II indicate that it would be possible to decrease the tip speed by 10 percent, increase the blade chord by 30 percent, and decrease the main rotor generated noise by 3.3 dBA.

And what would the penalty be? Increases in the weight of main rotor blade hub components, drive system (to accommodate the increase in torque), and increased tail rotor blade weight would be approximately 1 percent of gross weight (25 pounds) and approximately 0.5 percent in cost. If noise is important, the penalty is small, and in a competitive environment, it is invisible.

TABLE II. NOISE STUDY - "BEST" OF THREE ARBITRARY PLANFORM SHAPES NASA PO A65550B(66)

Parameter	Baseline	Wide Planform	Change
Tip Speed	665.7	599.1	-10%
Blade Chord	6.750	8.775	+30%
Planform	Constant	Constant	
Cruise Speed	126 Kn	126.5	
Vne	132 Kn	134	+1.5%
Hover Ceiling	11000	11700	+1.3%
Main Rotor P @ 90 Kn Cruise	141.5	131.5	-6.4P
Noise @ 90 Kn Cruise 500 ft	75 dBA	71.7	-3.3 dBA
Empty Weight	1150 lb	1173 lb	+2%
All Values Determined by Computer Program			

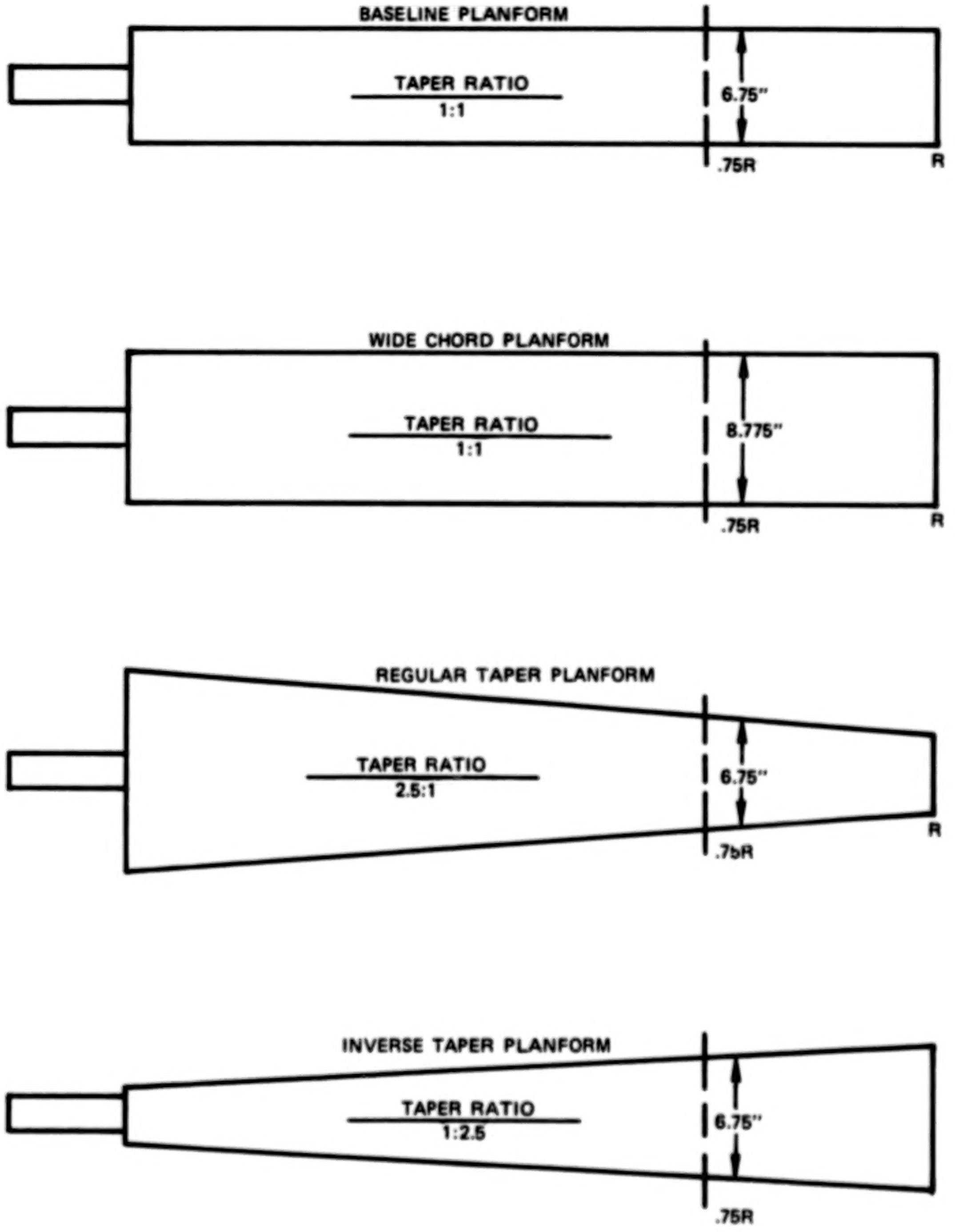
BASELINE PLANFORM
TAPER RATIO
1:1
6.75"
.75R
R
WIDE CHORD PLANFORM
TAPER RATIO
1:1
8.775"
.75R
R
REGULAR TAPER PLANFORM
TAPER RATIO
2.5:1
6.75"
.75R
R
INVERSE TAPER PLANFORM
TAPER RATIO
1:2.5
6.75"
.75R

Figure 2. Noise Study - Three Arbitrary Planform Shapes Compared to Existing Blades - NASA PO A65550B(66).

IMAGINATION AND SKILL

For the future, noise reduction potential is limited only by imagination and skill. Perhaps tail rotors will be unnecessary, but in any case, if designers all have the same constraints (by regulation and public demand) to produce quiet helicopters, the only real penalty will be to fall by the wayside if we do not.

We can look at:

Variable Speed Transmission

Variable Geometry Blades

Variable Geometry Powerplants with Improved Balancing Techniques

Improved "Designed-for-Noise" Structure

I cannot resist making some remarks about noise prediction. Much of this session has been dedicated to improved prediction capability for descriptors that do not effectively define the subjective reaction of the listener. I have a suggestion for a new technique, or at least I think it is new. Why don't we reproduce the "ears" of many listeners into one microprocessor with a switching mechanism from one ear to another. Recognition distance must be one of the requirements of the system. If the stay time of overhead is "very short," listeners will stop complaining. Considering that heart transplants are now routine, ear transplants should be easy.

It is going to be quiet, one way or another. If we do not insist on decreasing noise levels in our society, and not only in helicopters, then we will all lose part of our hearing capability. One way or another, it is going to be quiet.

SUMMARY OF WORK SESSIONS

FINDINGS, CONCLUSIONS, AND RECOMMENDATIONS

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SESSION I

H. K. Edenborough and J. M. Drees

INTRODUCTION

In establishing final recommendations from Session I, we first developed position statements relative to the subjects that had been covered in our session. We found that these fell into two logical divisions, the viewpoint of industry and that of government. In addition, statements relative to different facilities were developed and recommendations were then generated from these statement lists. The majority of these position statements and recommendations are self-explanatory. Others warrant additional discussion, which took place during the final session review. These comments and clarifications are included in the presentation of the statements.

POSITION STATEMENT

Industry Viewpoint

1. The industry is committed to reducing noise.

2. We are concerned that regulations may be enacted which are not compatible with available technology.

3. New regulations should include enough flexibility (including operational techniques) to preclude serious economic impact, especially with respect to derivatives.

4. We are impressed with the progress made in high speed impulsive noise technology.

5. We are less optimistic about the ability to achieve a full acoustic prediction capability within 5 years (e.g., in blade-vortex interaction and rotor-rotor interaction areas). There is concern that prediction techniques may lead to a global approach, not unlike the second-generation helicopter simulation program. Industry would feel more comfortable with the following approach:

(a) Identify all the modules required for a system prediction technique

(b) List and evaluate all available analyses (fundamental, empirical, and "semiempirical"). Select the best methods for application in the modules, consistent with the overall objective of the program use (consistent level of accuracy)

(c) Identify modules not having an adequate analysis; develop a method for it

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(d) Design a validated acoustic systems program using the modules selected to a level required for predesign and design evaluations within 4 to 5 years

6. The technical problem that must be addressed is an understanding of the source mechanisms for current helicopters (Bell models 222 and 412, Boeing Vertol 234, Hughes 500D, and Sikorsky S-76). These mechanisms include:

- Blade-vortex interactions
- Rotor-rotor interactions
- Broadband noise
- Engine core noise

7. Total system prediction techniques for climb, flyover, and approach must be developed to an accuracy of within +2 EPNdB in 4 to 5 years. Government should concentrate on this objective. (Also see the foregoing item 5(d).)

8. Validated acoustic wind tunnels and test techniques are not generally available. Imminent DNW (German-Dutch wind tunnel) results could be used for calibration of the National Full-Scale Facilities Complex (Ames 40- by 80-Foot/80- by 120-Foot Tunnel) and the Langley 4- by 7-Meter Tunnel, for instance. The industry consensus is that a new wind tunnel, specifically designed for low noise levels, should not be pursued. More extensive use of existing but acoustically treated facilities, such as the YO-3A aircraft and occasionally the DNW tunnel, is recommended.

9. Government should facilitate technology transfer by extending efforts beyond basic and generic research. It is important that U.S. industry be aware of the potential of emerging government-developed technology. The potential of such technology is most effectively evaluated by the principal researchers knowledgeable in the assumptions and subtleties of those methods. It is recommended, therefore, that government researchers:

(a) Clearly establish their current position on the technology accuracy

(b) Be alert to the need for timely assessment of their methods through an evaluation of parameters important to the industry

(c) Be prepared to educate and guide industry personnel in evaluating the new technology in as timely a manner as possible, i.e., before papers are published or new analysis are made available to the industry

While this recommendation to extend effort beyond basic and generic research runs contrary to present Administration policy, it is believed that NASA's involvement in solving some of industry's problems is necessary to add urgency and direction to NASA's basic and generic research. This will vitalize NASA's role in helping to maintain and improve the technology position of the U.S.

Government Viewpoint

1. There is more optimism and satisfaction with the state of understanding of some noise sources, such as compressibility (high-speed impulsive noise).

2. We are beginning to understand blade-vortex interaction noise.

3. Basic mechanisms of broadband noise are not yet understood.

4. Recent in-flight, wind tunnel, and hover room tests have provided new data for:

(a) Definition of scaling laws for high-speed impulsive noise and blade-vortex interaction noise

(b) Observation of shock propagation to far field

5. New analytical codes (3-D transonic) which are already being used for design studies of blade tips and are being intefaced with rotor acoustic prediction codes.

Facilities

Static

1. The Army Anechoic Rotor Hover Test Facility at Ames Research Center is excellent for small-scale hover studies.

2. No requirement was identified for a full-scale anechoic rotor facility.

Wind tunnels

3. Many good small-scale facilities are available for rotorcraft noise studies.

4. Medium and large-scale U.S. facilities are available but need significant noise treatment. This is currently scheduled for the Ames 40- by 80-ft wind tunnel and the Langley 4- by 7-m tunnel, and is being proposed for the 80- by 120-ft test section of the Ames National Full-Scale Facilities Complex (NFFC).

5. DNW is the best low-noise large wind tunnel currently available.

6. All wind tunnels must be carefully calibrated for acoustic and aerodynamic qualities and should be evaluated for rotorcraft noise studies.

Flight test

7. Using the YO-3A method is an excellent means of for gathering in-flight data. However, for rotor noise research higher speeds are required.

RECOMMENDATIONS

1. An effort should be made to provide an estimate of EPNL prediction accuracy using available acoustic prediction methods.

2. New regulations should include enough flexibility (i.e., operational techniques) to preclude serious economic impact and with special attention to aircraft derivatives.

3. Existing high speed impulsive noise technology should be incorporated into existing acoustic system prediction techniques.

4. A further understanding of acoustic source mechanisms for current generation helicopters (e.g., 222, 412, 234, 500D, S-76) should be developed and incorporated into acoustic system prediction techniques.

5. A concerted government/industry effort should be made to develop helicopter-system acoustic prediction techniques to within ±2 EPNdB within the next 5 years. An FAA representative noted that a new regulation may not be effective for 10 to 12 years. Meanwhile attention will be focused on operational aspects.

6. Existing wind tunnels should be acoustically calibrated and improved where necessary.

7. High-speed capability of in-flight acoustic measurement systems should be extended.

8. Government and industry should work together to form an effective means of acoustic technology transfer. The Army's acoustic research objectives differ from the civil and FAA requirements, yet there is community involvement because of Army aircraft flying over populated areas. Cooperation with NASA, however, will continue where possible.

9. Proposed American Helicopter Society and government plans should be integrated.

SESSION II

H. A. Morse and C. R. Cox

FLIGHT CONDITIONS

In attempting to organize and structure concerns and requirements for technology development, the flight conditions under which noise prediction improvements are most urgently required were listed and ordered in descending order of confidence in prediction methodology.

Descent Flight

Descent flight conditions are considered to be the most difficult for prediction of noise. This is believed to be true because it is the most complex fluid mechanics condition with the greatest rotor wake interaction. The near proximity or ingestion of the wake enhances the blade-vortex interaction (BVI) impulsive loadings. The turbulent wake also induces unsteady loading which results in multiple, directional noise sources combining into a moving noise generator with multiple directionality.

Level Flight

The second most difficult flight mode for noise prediction is the level flight flyby. The wake structure in level flight is not as close to the rotor plane as it is in descent, but it is now spread more due to the higher advance ratio. Due to the higher advancing blade Mach number, high-speed impulsive noise is of much greater significance, but in general the causes of noise generation, compressibility/transonic effects, BVI, and wake effects, are still a major concern. The degree to which these noise generation mechanisms are understood and predictable varies from fair to almost nothing. Even where predictability is feasible, complex and detailed techniques are required and more simple first approximation techniques are needed.

Takeoff

The takeoff or climb-out flight condition is less complex because of the tendency for the wake to be left behind. There are still uncertainties and problems in predicting the noise. The rotor is still a moving source with complex and highly directional properties. There is frequently a tendency to overpredict the noise for the climb-out condition. A postulated explanation is that hover is used as a reference and the rotor inflow is somewhat cleaner, while the wake is transported away from the rotor more rapidly in climbing flight.

Hover

Hover is, in practice, frequently unsteady due to light wind and turbulent flow ingestion. The near wake responds to unsteadiness and the resulting unsteady loads contribute to radiated noise.

TAIL ROTOR OR ANTITORQUE SYSTEM

The tail rotor noise prediction is complicated by all of the above noise source contributions and the interference and wake ingestion from the main rotor. The redeeming feature is that the antitorque system absorbs much less power than the main rotors, and weight penalties due to noise abatement techniques are less in terms of total payload.

SCALING AND PREDICTION

In working from noise prediction problem areas and perceived noise generation mechanisms, the need to address facilities and requirements became evident.

Current U. S. Wind Tunnel Facilities for Rotor Acoustic Testing

The U. S. has many excellent facilities and the technical expertise to maximize the productivity of these resources. Unfortunately none of the available U. S. facilities were designed and developed to adequately handle the unique requirements of rotorcraft acoustic testing. Some low cost improvements will enhance their capabilities to meet this requirement, but certain basic limitations will remain. One of the problems is lack of knowledge in the general area of model rotor acoustic testing. Available experience indicates that model rotor acoustic testing with 1/7-scale rotors is not only useful but essential for research and can be expected to be a requirement for any major departure from conventional design practices. Development of new rotor systems is very costly, and major advances will only be attempted if inexpensive model rotor tests indicate that low risk can be maintained. There is a short term requirement to maximize the use of existing facilities and a long term requirement for improved capability. Considerable uncertainty will remain in the ability to accurately predict the aerodynamically generated rotor noise of advanced geometry rotor systems.

Current Prediction Problem Priorities

The aerodynamically generated noise of main and tail rotors cannot be calculated in total from current techniques. There is a civil and military requirement for this capability and progress is being made by combined efforts of the research community. In the absence of fundamental knowledge, systematic procedures with appropriate scaling factors are essential. To accomplish this most efficiently, government and industry need to work together to isolate appropriate parameters, determine scaling factors, and validate the resulting component prediction equations to establish the range of usefulness and expected deviation of specific designs.

The ability to predict engine noise appears to be in relatively good shape. The engine manufacturers have developed appropriate noise predictive capability and the industry is satisfied with the present capability to estimate installation effects.

RECOMMENDATIONS

The session's deliberations led to the following short- and long-term recommendations. The short-term recommendations are aimed primarily at problem definition, while the long-term recommendations emphasize development and verification.

Short Term

1. Accurately evaluate and validate the capabilities of existing facilities to scale rotor acoustic emissions and define the future needs.

2. Compare selected past or new experiments with available analyses.

3. Blend the best analyses into a "Mark I" component system analysis.

4. Define the future analysis needs.

Long Term

1. Develop a "Mark II" component system analysis which includes selected aerodynamic and acoustic codes.

2. Expand and verify the fundamental analysis using testing facilities appropriate to noise source(s) under study. These facilities include:

   Available/improved wind tunnels
   Higher speed/climb inflight platform (YO-3)
   Anechoic hover chamber
   Detailed aerodynamic measurements on small- and full-scale instrumented rotors

OPERATING CONDITIONS

DESCENT
- BVI AND WAKE PROXIMITY
  - IMPULSIVE
  - BROADBAND

LEVEL FLIGHT
- IMPULSIVE (ADVANCING BLADE)
- UNSTEADY LOAD
  - TUBULENCE
  - BVI
  - WAKE GEOMETRY

TAKEOFF
- CONDITION UNIQUE, WITH LIMITED BVI

HOVER
- BLADE-VORTEX INTERACTION
- UNSTEADY FLOW

SCALING AND PREDICTION

CURRENT U. S. WIND TUNNEL FACILITIES FOR ROTOR ACOUSTIC TESTING
- ADEQUATE ONLY FOR SOME SOURCES
- REQUIRE IMPROVEMENTS
- INADEQUATE FOR ALL SOURCES AND MECHANISMS

LIMITED SCALING CORRELATION AVAILABLE DOWN TO 1/7 SCALE

SHORT-TERM REQUIREMENT TO USE AND EXPLOIT EXISTING FACILITIES

LONG-TERM REQUIREMENT FOR ACOUSTIC MODEL TESTING
WITH ANECHOIC FACILITY CAPABLE OF 1/6 SCALE (7-TO 10-FT DIA.)

CURRENT PREDICTION PROBLEM PRIORITIES:
- SOURCES
  - MAIN ROTOR
  - TAIL ROTOR
  - ENGINE (NOT MAJOR REQUIREMENT)

RECOMMENDATIONS

SHORT TERM (PROBLEM DEFINITION)

- ANALYSIS
  - DEFINE SELECTED EXPERIMENTS YOU CAN DO AND COMPARE WITH SELECTED ANALYSES
  - BLEND BEST ANALYSES INTO "MARK I" SYSTEM ANALYSIS
  - DEFINE LONG TERM ANALYSIS NEEDS
- SCALING
  - EVALUATE EXISTING FACILITIES
  - DEFINE LONG TERM NEEDS
  - VALIDATE SCALING CAPABILITIES

LONG TERM

- DEVELOP FUNDAMENTAL ANALYSIS
- VERIFY ANALYSIS
  - ACOUSTIC WIND TUNNEL DEVELOPMENT
  - DEVELOPMENT OF HIGHER SPEED/CLIMB IN-FLIGHT CAPABILITY (YO-3)
  - USE ANECHOIC HOVER CHAMBER
  - DETAILED AERODYNAMIC MEASUREMENTS

Session III

J. P. Raney and D. S. Jenney

This session included detailed discussion of analytical methods under development and general overviews of the state of the art of noise prediction. There are several important and encouraging activities in progress in the areas of acoustic and aerodynamic prediction for specific noise sources. Correlation of results with test data is being made parallel with predictions for these sources so that assessments of the methods are possible as they progress. These analyses of detailed mechanisms are not yet available in a total prediction system. Total aircraft noise is still predicted empirically, and the accuracy of that prediction is agreed to be inadequate.

The key conclusion of this session was that there is a need for an integrated system of noise prediction that uses the best available methods for each source and, at the same time, sums up the noise for comparison with system requirements (fig. 1). It seemed logical for NASA to take the lead in assembling this system analysis. The analysis would consist, initially, of identifying and prioritizing the key noise sources to be modelled. In performing this task opinions from government and industry would be carefully considered (fig. 2). The state of the art of each would be described. A responsible organization and individual would be identified to handle each source, select and describe the applicable method, and circulate for comment his assessment of the state of the art. As future research improves and replaces the individual source "modules," this assessment would be updated.

If NASA is to perform this function effectively it is clearly necessary for industry to provide its support in the form of available methodology, available data, assistance with state-of-the-art assessments, and general endorsement and acceptance of the effort (fig. 3). To make this a focused effort with a clear, bottom-line result, it is recommended that the system analysis be applied to a carefully selected problem or set of problems (fig. 4). Specifically, the analysis would be used to predict rotorcraft noise for the ICAO specification conditions in flyover, approach, and takeoff. The selected problems for evaluation of the methodology would be aircraft for which test data are, or will soon be, available. In this way, the bottom-line, total-system prediction can be judged. At the same time, the contributions of each module to the uncertainty or inaccuracy of the answer may also be judged. The result of this correlation will provide industry, NASA, and the certifying agencies with an audit of the state of the art, and will spotlight those elements of the process which contribute most to the residual error. Also, this activity will provide direction to future R&D programs.

In the selection of sample problems for correlation of the analysis, it is important to select at least two cases, one for which impulsive noise is a major contributor and one for which it is not. The uncertainty remaining, or the "state of the art " probably is different for those two cases, and in fact, there may be other classes which deserve similar separate attention (high engine

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noise case?). Further, the sample problems selected should be close to the proposed regulatory limit on noise; prediction accuracy is needed most when an aircraft is marginal relative to the ICAO levels, so aircraft grossly above or below those noise levels do not make good cases for calibration of the methodology.

There has been much discussion of the appropriate mechanization of a "system model." Prior examples such as Aircraft Noise Prediction Program (ANOPP) for CTOL acoustics and Second Generation Comprehensive Helicopter Analysis System (2GCHAS) for helicopter analysis are often cited as good or bad examples. At this state, the capability for predicting helicopter noise which is being described here is far less "integrated" than either of these. It is a "system" model in that it computes total aircraft noise, not just an element such as high-speed thickness noise. However, to do that it may combine a string of analytical and empirical modules or mechanism predictions. Initially, this string of modules would be those now available. The adequacy of these modules would be judged against the sample problems. As methods are upgraded, the modules would be replaced and, presumably, the bottom-line system noise prediction would improve.*

Fundamental aerodynamic tools are the basis of "first principles" noise prediction methods (fig.5). While the direction of the present program is correct, significant enhancements and extensions are required. In addition, the ability to calculate unsteady interaction effects must be developed. The present major development activities on transonic flow, wakes, and airloads on rotating blades should be continued with the objective of further refinement. The work on random airloads and viscous effects, including dynamic stall, unsteady boundary layers, and tip vortex formation and roll-up, needs a significant increase to yield fruitful results. A major new effort is required in the area of component unsteady aerodynamic interference, such as that between the main and tail rotor and between the main rotor and the fuselage. The aeroelastic effects of coupling aerodynamics and dynamics of rotors should also be investigated.

An aeroacoustic data base is required to improve the technology of rotorcraft noise prediction and reduction (fig. 6). This data base is intended to serve three purposes. First, it should be taken to answer specific questions about noise generating mechanisms or about the effectiveness of noise reduction techniques. Second, it should be used to generate empirical methods for noise prediction. Finally, it should be used to validate first principles predictions. Experiments which are "targets of opportunity," in the sense that aeroacoustic data may be obtained as an add-on to a test for other information, should be conducted when criteria assuring the quality of the aeroacoustic data may be met.

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*At the request of the industry representatives, a meeting was held on Thursday, April 1, 1982 at the NASA Langley Research Center to discuss industry's interest in having NASA develop an ANOPP-type noise prediction program for helicopters. A presentation was made by NASA personnel demonstrating the architecture and operation of ANOPP and the manner in which it can be used. Industry response of both acoustical staff members and engineering management was positive and indicated a desire for creation of a helicopter program similar to ANOPP and an intention to use it when developed.

There is a need to increase the utilization of available aeroacoustic test facilities and devices (fig. 7). No new facilities are contemplated at this time; however, improvements to and calibrations of those available are needed. The Ames 40- by 80-ft tunnel, the Langley 4- by 7-meter tunnel, and the Army Anechoic Hover Chamber are useful government facilities where greater utilization is desirable. One approach to increasing utilization is to rely on industry personnel to operate the facilities and for the design and fabrication of test models and hardware. The goal should be to increase utilization to a two-shift level.

The inflight measurement system typified by the YO-3 test should be developed to include better tracking of the sensor system, a large flight envelope, and higher flight speeds and rates of climb. This method is judged to be the best approach to obtaining free-field data on full-scale vehicles. If the YO-3 itself cannot logically be pushed to 180 knots or so, plans for an alternative aircraft should be initiated.

There is a need for an additional commitment of quality personnel to noise prediction and reduction research and development (fig. 8). This commitment must be made both by government and by industry. The personnel devoted to these activities should be involved in exchange plans to develop and improve perspective, encompassing both the research and design aspects of the helicopter noise problem.

CONCLUSION: INDUSTRY NEEDS AN INTEGRATED PROGRAM
FOR NOISE PREDICTION

RECOMMENDATION: NASA SHOULD DO THIS BEGINNING WITH
CURRENT CAPABILITIES

Figure 1

RECOMMENDATION: SOA ASSESSMENT BE MADE FOR EACH
CRITICAL METHODOLOGY

NOISE SOURCE	SOA METHOD	RESPONSIBLE CENTER
THICKNESS	REFERENCE	LaRC
BVI		
LOADING		
•		
•		
•		

Figure 2

ISSUE: IF NASA DOES THIS -

WHAT CONDITIONS SHOULD BE IMPOSED?

ANSWER: CLEAR INDUSTRY COOPERATION/ENDORSEMENT

Figure 3

RECOMMENDATION:

IDENTIFY PARTICULAR AIRCRAFT OR PROBLEMS
TO SERVE AS THE FOCUS AND TO QUANTIFY PROGRESS

PROPOSED:

- USE ICAO MEASUREMENT CONDITIONS
- SELECT TWO OR MORE HELICOPTERS, WITH AND WITHOUT SIGNIFICANT IMPULSIVE NOISE, FOR WHICH DATA DO OR WILL EXIST

Figure 4

AERODYNAMIC NEEDS

CONTINUED DEVELOPMENT

- MAJOR WORK TO DATE
  - TRANSONIC FLOWS
  - WAKES AND AIRLOADS
- MINOR WORK TO DATE
  - RANDOM AIRLOADS
  - VISCOUS EFFECTS
    - DYNAMIC STALL
    - UNSTEADY BOUNDARY LAYER
    - VORTEX FORMATION/ROLL-UP

NEW DEVELOPMENT

- MAIN ROTOR/TAIL ROTOR UNSTEADY INTERFERENCE
- MAIN ROTOR/FUSELAGE UNSTEADY INTERFERENCE
- AERODYNAMICS/DYNAMICS COUPLING

Figure 5

CONCLUSION: AEROACOUSTIC DATA BASE IS REQUIRED

1. ADDITIONAL DATA

   (A) ANSWER SPECIFIC QUESTIONS

   (B) VALIDATE METHODS

2. "TARGETS OF OPPORTUNITY" SHOULD BE SEIZED
   PROVIDED WE GET QUALITY DATA AND
   THE DATA WILL FILL A MEANINGFUL GAP

Figure 6

CONCLUSION: GREATER UTILIZATION OF FACILITIES IS NEEDED (E.G., TWO SHIFTS)

ALSO: ENLARGED FLIGHT ENVELOPE FOR YO-3 IS NEEDED

Figure 7

CONCLUSION: THERE IS A NEED FOR AN ADDITIONAL COMMITMENT OF QUALITY PERSONNEL TO NOISE PREDICTION/REDUCTION RESEARCH/DEVELOPMENT

Figure 8

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